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SATELLITE INTEGRATED POWER
AND ATTITUDE CONTROL
SYSTEM DESIGN STUDY

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SATELLITE INTEGRATED POWER AND ATTITUDE CONTROL
SYSTEM DESIGN STUDY

THESIS

Presented to the Faculty of the School of Engineering
of the Air Force Institute of Technology

Air University

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Requirements for the Degree of
Master of Science

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December, 1997

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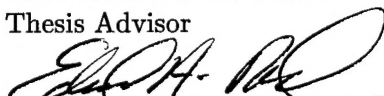

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Preface

There is a renewed interest in using flywheels as replacements for traditional chemical batteries in satellites. Using flywheels in an integrated system to replace not only the batteries but traditional attitude control systems as well, is of particular interest to satellite designers. This approach is referred to as an Integrated Power and Attitude Control System (IPACS).

Phillips Laboratory (PL) and NASA Lewis Research Center (LeRC) are currently conducting a cooperative research program on flywheels. There are several aspects to the system development and basic research and technology programs. The on-going system development programs include:

- An IPACS bench model to be built by SatCon, Inc. The objective is to evaluate the performance of a representative IPACS for Low Earth Orbit (LEO) satellites.
- A midsize Flywheel Energy Storage (FES) system by TRW and US Flywheel Systems (USFS).
- A small Open Core Rotator (OCR) flywheel being designed by FARE, Inc.
- Systems analysis that examines different aspects of flywheel technology. This includes research work done at the University of Maryland, Texas A & M, and Penn State, among others.

The *Satellite Integrated Power and Attitude Control System Design Study* is sponsored by Phillips Laboratory and is a part of the systems analysis program. The objective of the design study is to further the understanding of the use of flywheels in spacecraft design. The study compares the use of flywheels in satellite design to traditional design approaches for a variety of missions. This will allow for an assessment of the merits of flywheel technology in spacecraft design.

A computer model was built to accomplish this study. The model can serve as a tool for spacecraft designers or program managers to examine how the use of flywheels

may affect their programs. Phillips Laboratory plans to distribute this model to satellite system program offices and contractors in an effort to promote the use of flywheels. Many individuals in industry and government agencies have expressed interest in the model, including flywheel contractors such as SatCon, USFS, and FARE, Inc. have also expressed interest in the model.

We would like to acknowledge all the help we have received in the preparation of this thesis. First we would like to thank our advisors, Lt Col Kramer, Maj Pohl, Capt Agnes and Dr. Hall for their advice, guidance, and support. Our sponsor, Lt Chuck Donet at Phillips Laboratory provided us with valuable support. We would also like to thank certain people in government and industry who went out of their way to provide us with information and advice on flywheels and spacecraft systems. These include, but are not limited to:

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- Dr. James Kirk of FARE, Inc.
- Dave Christopher, Dave Hoffman and Susan Roalofs of LeRC
- Steve Ginter and Doug Havenhill of SatCon
- Chang Le of the SIBIRS Low Program Office
- Dr. Louis Kilmer of TECSTAR, Inc
- Kent Decker, Tom Pieronek and Doug Rusta of TRW
- Steve Bitterly of USFS

Finally, we would like to thank our wives (Jennifer, Kim, Michelle, and Trinette) and children for their support and understanding during our tenure at AFIT. It truly is a group project.

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List of Abbreviations

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| IPACS Integrated Power and Attitude Control System | iii |
| PL Phillips Laboratory | iii |
| LeRC NASA Lewis Research Center | iii |
| LEO Low Earth Orbit | iii |
| FES Flywheel Energy Storage | iii |
| USFS US Flywheels Systems | iii |
| OCR Open Core Rotator | iii |
| PACS Power and Attitude Control System | 1-1 |
| GEO Geosynchronous Earth Orbit | 1-2 |
| ACS Attitude Control System | 1-2 |
| CMG Control Moment Gyros | 1-2 |
| RPM Revolutions per Minute | 1-3 |
| SPO System Program Office | 1-4 |
| MWA Momentum Wheel Approach | 1-5 |
| GFA Gimbaled Flywheel Approach | 1-5 |
| HLM Hardware-Level Model | 2-7 |
| SLM System-Level Model | 2-7 |
| ESS Energy Storage System | 2-7 |
| PMAD Power Management and Distribution | 2-7 |
| EGS Energy Generation System | 2-7 |
| DET Direct Energy Transfer | 2-9 |
| PPT Peak Power Tracker | 2-9 |
| BCC Battery Charge Controller | 2-10 |
| PDU Power Distribution Unit | 2-10 |
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| TACA Traditional Attitude Control System Approach | 2-11 |
| Li-Ion Lithium Ion | 2-12 |
| BOL Beginning-of-Life | 2-15 |
| EOL End-of-Life | 2-15 |
| BDC Battery Discharge Controller | 2-16 |
| PCM Power Converter for Motoring | 2-16 |
| PCG Power Converter for Generating | 2-16 |
| CER Cost Estimating Relationships | 2-20 |
| FY92 \$k Thousands of Fiscal Year 1992 Dollars | 2-20 |
| NRCF Non-Recurring Cost Factor | 2-21 |
| RDT&E Research, Development, Test, and Evaluation | 2-21 |
| DOD Depth-of-Discharge | 2-22 |
| CDC Charge/Discharge Cycle | 2-22 |
| MEO Mid-Earth Orbit | 2-23 |
| NRCF Non-Recurring Cost Factor | 2-24 |
| RDT&E Research, Development, Test, and Evaluation | 2-24 |
| TACS/B Traditional Attitude Control System with Battery Energy Storage | 2-52 |
| TACS/FES Traditional Attitude Control System with Flywheel Energy Storage | 2-52 |
| IPACS/MWA Integrated Power and Attitude Control System/ Momentum Wheel Approach | 2-52 |
| IPACS/GFA Integrated Power and Attitude Control System/ Gimbaled Fly- wheel Approach | 2-52 |
| GMS GEO Meteorological Satellite | 2-61 |
| LIRS LEO Infrared Satellite | 2-61 |
| SBIRS-Low Space-Based Infrared Satellite—Lower Tier | 3-6 |

Abstract

Technological advances have brought Flywheel Energy Storage (FES) systems to the point where they can be used in satellites as replacements for chemical batteries. Flywheels have characteristics that may overcome limitations inherent in batteries. These characteristics include high specific energy, minimal degradation over time, and precise knowledge of charge level. A further advantage of flywheels is that, in a combined system, they can be used to replace traditional attitude control hardware. These Integrated Power and Attitude Control Systems (IPACS) show the potential to have less mass than the systems they replace.

The question this study seeks to answer is, "In what circumstances should FES be used on-board satellites and, if so, how?" The utility of power and attitude control systems using FES or IPACS was compared to baseline satellite designs using traditional battery energy storage.

The results of this study show that IPACS is generally superior to current technology batteries. IPACS is most effective in satellite applications with many charge/discharge cycles. In the case of geosynchronous Earth orbit missions batteries proved to be marginally superior in a few cases. But for low Earth orbit missions IPACS was shown to be markedly superior.

SATELLITE INTEGRATED POWER AND ATTITUDE CONTROL SYSTEM DESIGN STUDY

I. Introduction

Interest in incorporating flywheels into the power and attitude control system (PACS) on spacecraft dates back to the early 1960's. However, constraints in rotor and bearing technology prevented the use of flywheels as mechanical batteries. Recent advances in these flywheel-related technologies have moved the concept of using spinning wheels as mechanical batteries on spacecraft from the conceptual stage to reality. The purpose of this design study is to examine the use of flywheel energy storage (FES) and Integrated Power and Attitude Control Systems (IPACS) on board satellites.

1.1 Background

Earth-orbiting satellites have many subsystems that accomplish the basic functions required for spaceflight. The power subsystem and attitude determination and control subsystem are two of the major subsystems found on many spacecraft. The power subsystem accomplishes the functions of energy generation, energy storage, power management, and power distribution. The attitude determination and control subsystem orients the spacecraft, detecting and correcting errors in the attitude. Most Earth-orbiting satellites use solar arrays to generate primary electrical power. In some cases, a *primary* battery provides the primary power for a satellite. Primary batteries are intended for one time use—they cannot be recharged and are typically used for short duration missions. *Secondary* batteries are back-up power sources and are used when the primary power source cannot meet the power needs of the satellite. Secondary batteries are rechargeable batteries.

Satellite power demands and the number of charge/discharge cycles a battery must tolerate are both highly mission dependent. Orbit altitude and inclination determine the

number and length of the eclipses that satellites experience. For example, geosynchronous (GEO) satellites have two forty-five-day eclipse seasons per year, and one eclipse per day with a maximum eclipse duration of 72 minutes. In contrast, low-Earth orbit (LEO) satellites can have thousands of eclipses per year. The eclipses can be as frequent as one eclipse per orbit and have a maximum eclipse duration of 36 minutes (18). However, chemical batteries have certain disadvantages. For one, they have limitations on their depth of discharge and cycle life. It is not possible to repeatedly extract all of the energy from the battery, nor is it feasible to draw even a portion of the energy without limiting its lifetime. State of charge determination is another limitation. For some batteries, it is difficult to estimate precisely how much energy the battery contains. In addition, chemical batteries often require strictly controlled temperatures to optimize their performance. Together, these factors can lead to batteries being the mission-limiting factor on certain satellites.

Three-axis stabilized satellites require an attitude control system (ACS) to point the spacecraft in a desired orientation. Several methods are currently used to provide active three-axis attitude control. These include reaction wheels and control moment gyros (CMG). Reaction wheels and CMGs use low speed, metallic rotor flywheels. Reaction wheels provide attitude control by varying the rotation rate of flywheels configured in an array. Each reaction wheel is used to control the spacecraft angular momentum along a specific axis. Changing the rotation rates of the various wheels changes the angular momentum of the spacecraft. CMGs are similar to reaction wheels, but they are gimbaled. The wheels maintain a constant rotation rate, and using the gimbals to change the orientation of the rotors controls spacecraft attitude. Changing the orientation of the wheels changes the direction of the angular momentum vector. The orientation of the satellite changes in response to this angular momentum change.

1.1.1 Advantages of Flywheels as Energy Storage Devices. Advanced high-speed composite flywheels have many advantages over batteries as energy storage devices. Their most attractive feature is that they have a higher energy density than current chemical batteries. Also, they have less complex power conditioning requirements—batteries require controlled charging rates. Flywheels tolerate a broader

thermal environment—batteries have an optimal temperature range of about ten degrees whereas flywheels enjoy a 50 degree temperature range (30). These features can contribute to a lower spacecraft mass. When manufactured in quantity, flywheels are expected to cost less than batteries. Other anticipated advantages include a longer shelf life and no appreciable degradation over multiple charge/discharge cycles.

The greatest advantages of flywheels are realized when they are used in an IPACS. It is easy to imagine a system whereby the energy storage flywheels are used to control the orientation of a satellite in a manner similar to reaction wheels or CMGs. Because IPACS replaces two subsystems, it provides great mass and volume savings.

1.1.2 Limitations of Flywheels as Energy Storage Devices. The unknowns associated with using high-speed composite flywheels as satellite energy storage devices are significant. Energy storage flywheels are a new technology that have not yet flown in space. The low-speed metallic flywheels used for attitude control have provided some operational experience. However, these flywheels are much smaller in terms of energy storage capacity. They are generally constructed with metal rotors and physical ball bearings. They have low rotation rates on the order of a maximum of two to four thousand revolutions per minute (RPM). Energy storage flywheels will be constructed with composite rotors, use magnetic bearings, and have rotation rates on the order of tens to hundreds of thousands of RPM. Little is known about the fatigue lifetime and failure modes of composite rotors. The reliability of flywheels is yet to be fully understood. The most important advantages of flywheels for satellite program managers have yet to be proven. The risks associated with this new technology are high.

1.2 Importance of the Design Study

Phillips Laboratory, in cooperation with the NASA Lewis Research Center (LeRC), is currently investigating IPACS. In support of their research, we are investigating the use of flywheels both as energy storage devices and as a part of an IPACS. Many different organizations, companies, and universities have studied flywheels in space applications. Areas that have not yet been examined in detail include:

1. Can flywheels replace batteries for energy storage (FES)?
2. Can flywheels replace batteries and traditional attitude control systems in a combined energy storage and attitude control system (IPACS)?
3. When should FES systems or IPACS be used?
4. When using IPACS, should the configuration include the momentum wheel approach or the gimbaled flywheel approach?
5. Should flywheels be spinning at launch or should they be locked?
6. What flywheel characteristics have the most effect on the performance of flywheel-based systems?
7. What are areas of significance for future research?

Phillips Laboratory hopes to analyze these questions with an integrated software model. Their goal is to acquire a tool where a user can enter the specifications of their satellite, including the mission profile, chemical battery data, and flywheel data. The model will then compare various design configurations using flywheels and batteries and report back to the designer which configuration is the best for that scenario. Phillips Laboratory would like to use this model to answer questions from the satellite System Program Offices (SPO) and contractors about the effectiveness and advantages of flywheels.

1.3 Statement of the Problem

Given the desires of our sponsor, our problem statement is as follows:

Examine satellite flywheel-based energy storage and integrated attitude control systems. Compare these flywheel-based systems to traditional systems and determine if, when, and in what configuration energy storage flywheels can be effectively applied in satellite design.

1.4 Objectives of the Design Study

The basic objective of this design study is to investigate the use of flywheels in satellites. We focused our efforts to answer the following questions:

- When should a FES system or IPACS be used?
- When using IPACS, should the configuration include the momentum wheel approach (MWA) or the gimbaled flywheel approach (GFA)?
- Should flywheels be spinning at launch or should they be locked?
- What flywheel characteristics have the most influence on the performance of flywheel-based systems?
- What are significant areas for future research?

1.5 Thesis Overview

Our thesis is organized into four chapters and three appendices. Chapter 1 begins with an introduction to the design study and thesis. Chapter 2 presents our design study methodology. It discusses the objective of our methodology, the systems engineering process we used, how we modeled the problem, and our analysis approach. Chapter 3 discusses the analysis of our data. Finally, Chapter 4 presents our findings and conclusions.

The appendices discuss the processes we used to conduct the design study and our research. Appendix A presents how we accomplished the initial system study. Appendix B discusses how we performed the second, more detailed system study. Appendix C contains our database of annotated references. Following the appendices are the bibliography and vitas of the authors.

II. Methodology

This chapter describes the processes used to accomplish our design study. It explains the systems engineering framework that drove our design efforts, our method of modeling the important features of the design problem, and our approach for determining the best alternative designs using flywheels for attitude control and energy storage. We begin with systems engineering.

2.1 Systems Engineering

Systems engineering is a discipline whose scope makes definition a difficult task—description is a more tractable pursuit. It encompasses problem definition, characterization of the decision-maker, identification of alternatives, evaluation of alternatives against the model of the decision-maker, optimization, and recommendations for resource allocation. It is characterized by a structured process for accomplishing these tasks. The process is iterative, running from the initial identification of a need for a system to the final disposition of the system. This iteration can follow the waterfall approach typical of manufacturing processes, or it can be a spiral with periodic refinement of the design. The exact structure of the process is driven by the nature of the problem and the intuition of the system engineer.

2.1.1 The Classical Approach. Our initial application of systems engineering was based on the method developed by Hall (9). This approach divides systems engineering into an iterative process comprising seven steps. The steps are:

1. Problem Definition
2. Value System Design
3. System Synthesis
4. System Analysis
5. Modeling and Optimization

6. Decision Making

7. Planning for Action

The method has been in use for almost 40 years and has been extensively modified and specialized by various practitioners (13), (21), (28), (29). We incorporated some of these modifications into our approach. Specifically, we used nine of the twelve products of problem definition from Hill and Warfield's expansion of Hall's method (13). We also used a value system design based on information presented in Sage's text (29).

When we applied this method to our system design problem we encountered several problems. First, the ambiguity of some of the elements of Hall's method needed further refinement. The three steps, *System Synthesis*, *System Analysis*, and *Modeling and Optimization*, tend to run together. This blurring is apparent in the literature as many practitioners have redefined these steps in more meaningful ways (28). We made an initial determination of what would be accomplished in each of these steps and began to apply Hall's method in earnest. Our next problem was one of knowledge. We found ourselves trying to define our system design problem when we knew very little about the underlying physical and engineering processes. We made some guesses as to what high-level design factors would be important as well as some estimates about what the environment and system scope should be. We moved on to the value system design step and found ourselves in a similar situation. We built an objective hierarchy around costs and performance measures, but lacked a quantitative link between system performance and objective measures. While all of this activity was in progress, we were also conducting research. Hall's method did not provide a clear procedure for incorporating new information. We backed up and revised our problem definition and value system several times—without performing any modeling or assessment of our initial set of alterables. As a result, we lacked a clear picture as to what difference the new information made. When we reached system synthesis, our information and problem structure were still at a very high level. We generated a list of possible systems using a strategy table and the alterables we identified in the problem definition step. Moving into system analysis led to the next problem—we did not have sufficient information to quantitatively analyze the alternative designs. We attempted to optimize our design recommendation using a quantitative quality function

Table 2.1 Model-Based Approach vs. Hall's Method

| Model-Based Method | Hall's Method |
|-----------------------|--|
| Problem Definition | Problem Definition |
| Model Definition | Value System Design System Synthesis System Analysis |
| Modeling and Analysis | Modeling and Optimization |
| Decision Making | Decision Making |
| Implementation | Planning for Action |

deployment process. We modeled each alternative using subjective evaluations of alterable performance in each objective measure. Our results led to a decision to proceed to the next iteration without eliminating any of the alternatives. We decided to review other system engineering methods and see if they might be more applicable to our design problem.

2.1.2 Integrated Modeling and Object-Oriented Design. Our review of other system engineering processes uncovered some more modern system engineering methods. These methods make use of computers and models to capture important aspects of the problem. We continued to express our design progress in terms of Hall's method while running a parallel, model-based design approach. This section discusses how we implemented this approach.

2.1.2.1 Integrated Modeling.

The Model-Based Approach. The major problem we encountered in applying Hall's method was that there was no central tool for communicating the state of our knowledge of the problem. As a result, our efforts were disjointed and somewhat ad hoc. We addressed this problem by moving to an integrated modeling approach. This effectively resulted in incorporating modeling into every step of Hall's process. Hall's process was kept for two reasons. First, it provided a familiar tool for communicating results to those not familiar with model-based techniques. Second, we developed our model-based approach as we applied it and Hall's method provided a yardstick to measure its completeness. Table 2.1 shows how our final process compares to Hall's method.

Each of these steps is discussed below.

The problem definition step is similar to the problem definition step in Hall's method, except that it is expanded to include the value system design step. The purpose of the problem definition step is to identify the characteristics of the problem to include:

1. A clear statement of purpose
2. An initial assessment of scope
3. An initial assessment of decision-maker requirements
4. An initial assessment of decision-maker utility
5. An initial assessment of decision-maker values

These characteristics are used to create models of the system, the decision-maker, and the environment. As new information becomes available, the initial assessments can be revised to reflect the new information. This additional information is then communicated throughout the system engineering process when it is incorporated into the model.

The model definition step has two parts: application domain modeling and computer domain modeling. Application domain modeling is concerned with describing the model's target in terms of those real-world characteristics that are important to the decision. Computer domain modeling translates the application domain model into a computer model. For example, we modeled flywheels. The application domain is the physical world. We determined which physical characteristics of flywheels would be important to our decisions. We then created a mathematical model of the flywheel using real-world information to determine the values associated with these characteristics. This is the application domain model. We then constructed a computer model of a flywheel by translating the mathematical description of the application domain model into its computer analog. The computer model can then be used as a surrogate for the system under study. We used this technique to create models of all the important aspects of the system design.

The modeling and analysis step is used to evaluate the current state of the model and to generate decision recommendations. During this phase the model is used as a surrogate for a real system. Alternative system configurations are evaluated and ranked. Point

designs are optimized using the decision-maker model. When a clear answer emerges, the results are passed to the decision-maker.

Decision-making is the same as in Hall's method. This is where the Decision-maker examines the recommendation and makes the decision.

Implementation is carrying out the decision. It includes all of the planning actions associated with the decision, as well as the execution of the decision.

This approach has several key features:

1. All requisite knowledge of the problem is explicitly modeled.
2. The approach uses internal iteration to constantly update the model to reflect the state of knowledge.
3. Once the initial model is created, it is always possible to provide the decision-maker with recommendations.
4. Changes and what-ifs can be examined for importance without changing the current model.
5. The decision-maker's values and preferences are explicitly modeled and easily modified.
6. The model can be used as a communication tool and for interface control when integrating inputs from specialty engineers.
7. Important changes can be implemented quickly.

Application of the Model-Based Approach. We applied the model-based approach in parallel with our second iteration of Hall's method. We created models of the requirements, the system, and the decision-maker and iteratively updated the model until we felt we had enough information to perform an analysis and make a design recommendation. As before, when we tried to translate our progress into the Hall framework, we had difficulty expressing just what tasks were accomplished in system synthesis, analysis, and modeling. Also, the internal refinements to the model were happening

too fast to track using Hall's method. We accomplished, in a single pass through the design process, the equivalent of moving from the coarse to the hyperfine structure in Hall's method.

2.1.2.2 Object-Oriented Design. A model-based approach requires a model. The method for constructing the model is not defined in the approach. There are several modeling approaches available. We considered functional decomposition/functional allocation and object modeling techniques. These two types of approaches examine design from opposite directions. Either can be used to design a system, but each has strengths that can be exploited for different types of problems. Functional decomposition techniques first identify tasks that need to be accomplished and then specify the agent(s) that will accomplish the task. These techniques are appropriate for conceptual systems or for new systems, where the methods for accomplishing the functions are potentially not known or understood. Object modeling techniques are appropriate when the problem can be decomposed into classes of objects. These classes are established based on common data and functions. Since our problem consisted of accomplishing defined functions with defined hardware, an object oriented approach was adopted. The next section describes how we implemented our model.

2.2 Modeling

2.2.1 Overview.

Purpose of Modeling. A perfect "magic" database containing flywheel performance, battery performance, and system integration data would be the easiest and most accurate reference to obtain the answers to our research questions. Since such a single source of information did not exist, we had to resort to producing the answers ourselves. A major portion of our thesis effort was the development of a high-level model. The resulting output assisted in a clear mapping of our design space. The model's primary purpose was to analyze the various PACS hardware combinations for the best alternative.

Basic Structure. The model is divided into two major components, the hardware-level models and the system-level model. The hardware-level Models (HLMs) are the first key part of our effort. These models simulate the performance of the basic satellite hardware, yet it is the functional area of the hardware that determines the individual model boundaries. Each model is composed of the hardware that performs a specific class of functions.

The second major component is the system-level model (SLM). The system-level model performs the analysis on the aggregate of the hardware-level models. It generates satellite PACS alternatives by combining all possible mixtures of hardware elements. This model controls the simulation from the top of our system hierarchy.

2.2.2 Description of Hardware. We examined six classes of spacecraft components that fall into four functional areas. These functional areas are the energy storage system (ESS), power management and distribution (PMAD), energy generation system (EGS), and attitude control system (ACS). Under our hardware-level model, the physical components are very important. This section describes the hardware components included in the scope of our model.

Flywheels A flywheel is defined as a rotating mass used to store energy in mechanical form. A FES system consists of a flywheel rotor, in the shape of a cylinder or disk, made from composite material. It spins at an extremely high speed and stores rotational kinetic energy. Energy is added to the flywheel using an electric motor to place a torque on the rotor. The result is an increase in flywheel rotational speed and therefore, an increase in rotational kinetic energy. Electric energy is extracted from the flywheel using a generator driven by the decelerating wheel. The same piece of hardware serves both as the motor and generator.

In order to reduce energy lost to friction, the motor/generator is brushless and the rotor is spun on magnetic bearings. Physical or ball bearings create unacceptable energy losses. They are only used during launch (if applicable) or as touchdown surfaces in the case of magnetic bearing failure. Most flywheels use active magnetic bearings to maintain the rotor in a desired position. Sensors provide position and

orientation information to a controller. Generally two sets of bearings are used along the spin axis. A third bearing is mounted perpendicular to the other two. The power consumption of the bearings, sensors, and controllers are accounted for as part of the spacecraft housekeeping power requirements.

Rotors are constructed of composite materials so they can be spun at higher speeds and achieve a higher energy density than traditional satellite reaction wheels. There are many different approaches to the configuration of flywheels. Some use rotors that are disk shaped while others use cylinders. Some designs use rotors that are mounted on a shaft to which the magnetic bearings and motor/generator are also connected. FARE, Inc. has an open core, shaftless rotor design.

All the various flywheel designs share certain traits. They have rotation speeds on the order of tens of thousands of RPMs, high specific energies, and flexibility in power management. Flywheels can be sized to specific energy storage requirements by varying its size, maximum speed, motor/generator, and the desired power transfer rates.

Flywheels intended for terrestrial use are normally fitted with a containment system to safely hold a failed rotor. These containment systems are quite heavy and, if used for space-based flywheels, render the flywheels impractical for use (26). The approach to safety used for space-based flywheels is to derate the maximum operating speed of the rotor to the point where there is sufficient confidence that it will not fail during the flywheel's design life. Sensors are also available to monitor the rotor and provide advance warning of any impending failure in sufficient time to avert the failure.

Flywheels can be designed to be launch-rated to provide power during launch. There are two basic changes required to make a flywheel launch rated—a pressure vessel and stronger bearings. The pressure vessel is used to maintain a vacuum, which eliminates atmospheric drag on the rotor during operations on the pad and during launch. The bearings need to be stronger so that they can withstand the forces and vibration environment during launch. This can be accomplished with larger magnetic bearings or physical bearings that are retracted once the satellite is in orbit.

Secondary Chemical Batteries Rechargeable chemical batteries are currently the most commonly used method for Earth-orbiting spacecraft energy storage. , During the sunlit portion of the satellite's orbit the batteries store energy generated by the solar array through a reversible chemical reaction. Batteries provide energy during the eclipse portion of the orbit.

There are several different types of secondary batteries that possess the high energy density and cycle capacity required for Earth-orbiting satellite missions. The two most prevalent types are Nickel-Hydrogen and Nickel-Cadmium batteries. Lithium Ion is a relatively new battery technology expected to outperform the others in the future. Lithium-Ion batteries provide very high energy densities and are envisioned as the next generation of primary batteries for space applications.

Primary Chemical Batteries Primary chemical batteries also store energy using chemical combinations. Unlike secondary batteries, primary batteries serve as the primary power source for a spacecraft. They are not rechargeable, and are mainly used for one-time applications such as powering satellites from launch through the deployment of the solar array. These batteries can be designed with much higher energy densities than secondary batteries. Space missions use various types of primary batteries.

Power Management and Distribution The power management and distribution system forms the nexus of the ESS, EGS, and the spacecraft loads. Controlling the solar array output, the storage of energy, and the regulation of the spacecraft electrical bus voltage are what define the power management function. The power distribution function provides switching and transmission of power from the electrical bus, including activating and switching off individual spacecraft loads. The primary components are regulators, power converters, power distribution units, and wiring harnesses.

The means by which the solar array output is regulated while the spacecraft is in sunlight distinguishes the type of PMAD system. Typical architectures include the direct energy transfer (DET) method and an approach that utilizes a peak power tracker (PPT). The first uses shunt regulators in parallel with the solar array to regulate the output. It accomplishes this by dissipating excess energy during sunlight.

The second uses a dc-dc converter in series with the solar array to actively change the operating point of the solar array up to the peak output power.

For a battery ESS it is usually necessary to provide independent Battery Charge Controllers (BCC) at the battery input in order to stress the batteries as little as possible, thus slowing the rate of degradation. Also, since the output voltage changes during battery discharge, there is a need for output voltage regulators to maintain the spacecraft bus voltage.

If the spacecraft bus voltage is below the upper flywheel voltage, the ESS will require power converters at the flywheel input. It is assumed in this study that the spacecraft bus is 28 Volts dc, though a spacecraft with a higher voltage bus may be better suited for the utilization of a flywheel ESS. . Also, since the output voltage changes during flywheel discharge there will be a need for output voltage regulators to maintain the spacecraft bus voltage.

All PMAD systems include a power distribution unit (PDU) to provide for the transmission and switching of power from the spacecraft electrical bus. Rarely are spacecraft loads hard-wired to the electrical bus. The PDU provides for the availability and discontinuation of power to the spacecraft loads, as well as fault tolerance for survivability should an anomaly occur.

The final components are the wiring harnesses. These components act as conduits and route electrical power between the various PMAD system components.

Energy Generation A spacecraft requires energy to power its payload and supporting subsystems. Designers use an energy generation subsystem to produce energy throughout the satellite's lifetime.

The most common energy source used on Earth-orbiting spacecraft, photovoltaic solar cells, convert incident solar radiation directly to electrical energy (16). Solar cells connected in series and parallel combinations comprise a solar array. Its configuration is either planar or concentrator, and either type can be body- or panel-mounted. Panel-mounted solar arrays are usually found on three-axis stabilized spacecraft, and

are mounted on a boom. The body-mounted approach reduces the requirements for tracking and pointing on the spacecraft and is typically used on spinning spacecraft.

Attitude Control The attitude control system controls the spacecraft pointing. There are a number of different methods of performing attitude control depending on the spacecraft mission requirements. For this analysis, Earth-pointing was assumed, along with a need for high pointing accuracy to less than 1 degree error. Also included was the possibility of changing spacecraft attitude by slewing the spacecraft. As a result, a system with spinning wheels can be assumed. There are then two choices: use an existing ACS or use the flywheels for attitude control. An existing system is henceforth referred to as a traditional attitude control system (TACS) approach (TACA). Reaction wheels, momentum wheels, and control moment gyros all fit into this category. TACS controls attitude only.

The second choice is referred to as an integrated power and attitude control system (IPACS). Under this category are two choices, momentum wheel approach (MWA) and gimbale flywheel approach (GFA). An IPACS/MWA operates by changing flywheel speeds to store and deliver energy and to control attitude. An IPACS/GFA operated by changing flywheel speeds to store and deliver energy and by changing the spin axis of the flywheel with respect to the rest of the spacecraft to control attitude. The only parts of the ACS within the scope of this study are the flywheels and, if present, the gimbaling mechanism.

2.2.3 Hardware-Level Modeling. The requisite features of the hardware we have just described were modeled to produce the lowest tier of our PACS model. This section provides a detailed look at the assumptions, data, and mathematical models that we used.

2.2.3.1 Assumptions. The hardware-based models made the following assumptions to effectively model the different alternatives.

- The spacecraft is in a circular orbit.

- The worst-case external torque is a single torque equal to the sum of all the external torques acting perpendicular to the spacecraft's angular momentum vector.
- External torques can be generalized over an orbit into secular or cyclical.
- The ACS components that can be varied are the flywheels, the addition of a gimbal unit, and the addition of a separate set of attitude control reaction wheels or control moment gyros.
- Other ACS components, such as electronics, thrusters, and sensors that are common in all ACS alternatives are ignored for calculation of performance measures.
- All flywheels used in a single alternative were identical.
- The flywheels have an upper and lower rated rotation rate.
- Flywheels experience no loss of performance/capacity over time. Flywheel rotors are designed such that they can be operated to the same maximum speed for their entire life without failure due to fatigue.
- The average efficiencies for the motor and generator remain constant over time.
- The average power loads for the magnetic bearings and controller remain constant over time.
- Other small losses and loads are ignored.
- Battery and satellite bus voltages are nominally 28 V.
- Lithium Ion (Li-Ion) batteries are only applicable for missions requiring fewer than 2000 charge/discharge. Predictions for Lithium-Ion technology estimate the maximum possible number of charge/discharge cycles to reach 2000 in the near term. (4)
- The spacecraft bus is fully regulated.
- The wiring efficiencies are 99% between the solar arrays and PMAD components, and 99.5% between all other components.
- The power distribution unit (PDU) efficiency is the same for all architectures.
- The PPT efficiency is the same for all architectures that incorporate it.
- The end-of-life PMAD system reliability is 99%.

- The model assumes that solar arrays provide the energy generation function. Solar arrays are the predominate form of energy generation on-board Earth-orbiting spacecraft.
- The temperature of the solar array decreases linearly as the altitude of the spacecraft increases from LEO to GEO. Larson (16) provides typical temperatures at both LEO and GEO altitudes.

2.2.3.2 Hardware Model Inputs. The following inputs are used to create the required output and performance measures.

Table 2.2 Inputs - Hardware Data

| Name | Units | Description | Functional Area (FA) |
|------------------------|-------------------|--|----------------------|
| P_{Wheel} | W | Flywheel housekeeping power | PMAD |
| T_{taca} | Nm | The peak torque that the separate ACS is capable of providing the spacecraft | ACS |
| P_{taca} | W | The power necessary to operate the separate ACS | ACS |
| $ArraySP$ | W/kg | Specific power of selected array material | EGS |
| $AnnDegs_A$ | | Annual degradation of selected array material | EGS |
| $MaxAngInc$ | deg | Maximum angle of incidence on solar array | EGS |
| I_s | kg m ² | Inertia of the rotor and any components spinning with it | ESS ACS |
| ω_{max} | rad/s | Maximum operating rotational speed of the rotor | ESS ACS |
| ω_{min} | rad/s | Minimum operating rotational speed of the rotor | ESS ACS |
| $Controller Power$ | W | Average power consumption of flywheel controller | ESS |
| $Bearing Power$ | W | Average power consumption of the magnetic bearings | ESS |
| $FlyMass$ | kg | The mass of one flywheel | ESS |
| $FlyVolume$ | m ³ | The volume of one flywheel | ESS ACS |
| $FlyMass_{LR}$ | kg | The additional mass incurred in a launch-rated flywheel | ESS |
| $FlyVol_{LR}$ | m ³ | The volume delta for a flywheel rated for launch | ESS |
| $Flywheel Reliability$ | | Probability a flywheel will be functional at end-of-life | ESS |
| $Battery Type$ | | Type of battery (Ni-Cd, Ni-H ₂ , or Li-Ion) | ESS |
| $Rated Capacity$ | Ahr | Rated capacity of each battery | ESS |
| $Battery Mass$ | kg | Unit mass of each battery | ESS |
| $Battery Volume$ | m ³ | Unit volume of each battery | ESS |
| T_{mg} | Nm | The peak torque that the motor is capable of delivering to the rotor | ACS |
| Res | % | The accuracy to which the wheel speed can be controlled | ACS |

Table 2.3 Calculations Passed Between Hardware Models

| Name | Units | Description | FA |
|-----------------|-------|--|-------------|
| E_{Slew} | Whr | Energy for slew | PMAD |
| E_{GEO} | Whr | ACS GEO Orbit Energy | PMAD |
| P_{ACS} | W | ACS operating power | PMAD |
| P_{SA} | W | Power provided by the solar arrays at the end-of-life | ACS |
| $P_{EG,BOL}$ | W | Total solar array output power at beginning-of-life (BOL) for a Flywheel ESS | PMAD |
| $P_{ES,BOL}$ | W | The portion of solar array output power required to store energy at BOL | PMAD |
| P_{OpBatt} | W | Operating power PMAD needs from EGS for loads at end-of-life (EOL), given batteries | EGS |
| $P_{RpEOLBatt}$ | W | Recharge power PMAD needs from EGS to recharge batteries at EOL | EGS |
| $P_{TotEOLFly}$ | W | Total power PMAD needs from EGS at EOL, given flywheels | EGS |
| T_{Orb} | min | The time it takes the satellite to complete one orbit | PMAD ACS |
| T_{MaxEcl} | min | Maximum amount of time that the satellite is in darkness | PMAD |
| Ecl_{max} | | | ACS |
| E_{MaxEcl} | Whr | The total energy (integrated power) required at the load input over the maximum eclipse duration | PMAD |
| N_{Batt} | | The number of batteries needed to satisfy the energy storage requirement | PMAD |
| N_{Fly} | | The number of flywheels needed to satisfy the energy storage and attitude control requirements | PMAD ACS |

Table 2.4 Inputs - Efficiencies

| Name | Units | Description | FA |
|-----------------|------------------------|---|------|
| η_{PPT} | | PPT efficiency which accounts for the PPT accuracy and series regulation losses | PMAD |
| η_{PDU} | | PDU efficiency which accounts for losses associated with power flow through the PDU | PMAD |
| η_{BCC} | | Battery charge controller (BCC) efficiency which accounts for losses associated with power flow through the BCC during battery charging | PMAD |
| η_{BDC} | | Battery discharge controller (BDC) efficiency which accounts for losses associated with power flow through the BDC during battery discharge | PMAD |
| η_{BC} | | Battery charge efficiency which accounts for the losses associated with the battery's ability to store energy | PMAD |
| η_{BD} | | Battery discharge efficiency which accounts for the losses associated with the battery's ability to provide energy | PMAD |
| $ArrayEff$ | | Efficiency of selected array material | EGS |
| η_{TempSA} | deg C ⁻¹ | Temp inefficiency of selected array material (per deg C above 28 deg C) | EGS |
| η_{PCM} | | Power converter for motoring (PCM) efficiency which accounts for losses associated with power flow through the power converter that conditions the input power to the flywheel motor | PMAD |
| η_{PCG} | | Power converter for generating (PCG) efficiency which accounts for losses associated with power flow through the power converter that conditions the output power from the flywheel generator | PMAD |
| η_{Mot} | | Motor efficiency which accounts for the losses associated with the flywheel motor's ability to store energy in the flywheel | PMAD |
| η_{Gen} | | Generator efficiency which accounts for the losses associated with the flywheel generator's ability to provide energy | PMAD |
| η_{EGS-ES} | | Energy generation to energy storage (flywheel) efficiency | ACS |
| η_{DesSA} | | Efficiency associated with design of solar array | EGS |
| η_{ES-L} | | Energy storage (flywheel) to loads efficiency | ACS |
| η_{EGS-L} | | Energy generation to loads efficiency | ACS |
| η_{IW} | | Flywheel to flywheel efficiency | ACS |

Table 2.5 Inputs - Mission Data

| Name | Units | Description | FA |
|--------------------|-------------------|--|----------------|
| <i>Design Life</i> | years | Mission design life requirement | EGS ESS ACS |
| M_{Dry} | kg | The mass of the payload and spacecraft systems (not including propellant mass) | PMAD |
| I_x, I_y, I_z | kg m ² | The principal moments of inertia of the spacecraft | ACS |
| A | m ² | The maximum cross sectional area of the spacecraft | ACS |
| <i>ArrayCfg</i> | | Array configuration (Panel-mounted, body-mounted/ cylindrical, or body-mounted/ omni) | EGS |
| $PMAD_{Type}$ | | Determines whether to model a DET based system or a PPT based system | PMAD |
| t_{slew} | minutes | The minimum expected duration for the slew maneuver | ACS |
| N_{slews} | | The expected number of slew maneuvers per orbit | ACS |
| θ_{slew} | degrees | The maximum expected angular displacement of a slew maneuver | ACS |
| P_n | W | The power requirements of the load over the orbit, broken up into a maximum of 5 discrete power levels and durations | PMAD ACS |
| LD_n | minutes | The duration of the associated power requirement | ACS |
| P_{Min} | W | The lowest non-zero load from the load profile | PMAD |
| <i>Altitude</i> | km | Altitude for selected mission | EGS ESS ACS |
| θ_{maxdev} | degrees | Maximum deviation from Earth-pointing allowed | ACS |

2.2.3.3 Constants and Assumed Values. The following values are either constants or assumed values used in the calculations. The values listed are from Larson (16) and Agrawal (1). Drag coefficient is a worst-case assumption. Typical drag coefficients are approximately 2. The inclination factor is chosen as the worst-case value, as the usual value changes over an orbit, limited to between 1 and 2. Other values include typical values for spacecraft.

Table 2.6 Constants

| Name | Value | Description | FA |
|----------------|---|--|-----|
| C_d | 4 | Drag coefficient | ACS |
| $c_{pa} - c_g$ | 0.1 m | Distance between the center of atmospheric pressure and the center of gravity of the spacecraft | ACS |
| $c_{ps} - c_g$ | 0.1 m | Distance between the center of solar pressure and the center of gravity of the spacecraft | ACS |
| μ | $398601 \text{ km}^3/\text{s}^2$ | Gravitational constant for Earth | ACS |
| D | 1 amp-turn-m^2 | Residual magnetic dipole of the spacecraft | ACS |
| M | $7.96 \times 10^{15} \text{ m}^3 \text{ Tesla}$ | Magnetic moment of the Earth | ACS |
| F_i | 2 | Inclination factor accounts for the magnetic field changing with respect to the magnetic inclination | ACS |
| K | 1358 W/m^2 | Average intensity of the solar radiation at Earth | ACS |
| $MinSInt$ | 1309 W/m^2 | Minimum solar intensity at aphelion | EGS |
| q | 0.6 | Reflectivity of the spacecraft | ACS |
| c | $3 \times 10^8 \text{ m/s}$ | Speed of light | ACS |
| Bus Voltage | 28 V | Bus voltage | ESS |

2.2.3.4 Hardware Model Outputs.

The functional areas create the following outputs.

Table 2.7 Outputs

| Name | Units | Description | FA |
|----------------------|----------------|--|---------------------|
| E_{Req} | Whr | The total amount of energy that must be stored while the satellite is in sunlight, given a battery ESS | PMAD |
| $P_{ES,EOL}$ | W | The portion of solar array output power required to store energy in the batteries at EOL | PMAD |
| P_{Op} | W | The portion of solar array output power required to support the loads, given a battery ESS | PMAD |
| E_{Req} | Whr | The total amount of energy that must be stored in the flywheels while the satellite is in sunlight | PMAD |
| $P_{EG,EOL}$ | W | Total solar array output power at EOL for a Flywheel ESS | PMAD |
| η_{Fly} | | The total efficiency for power transfer between flywheels | PMAD |
| η_{EG-ES} | | Total efficiency for power transfer from the solar array output to the flywheel input | PMAD |
| η_{EG-L} | | Total efficiency for power transfer from the solar array output to the load input | PMAD |
| $FlyCapacity$ | Whr | The energy storage capacity of a flywheel | ESS |
| $FlyPower$ | W | The average housekeeping power required by a flywheel | ESS |
| $Effective Capacity$ | Whr | The effective energy storage capacity of the battery | ESS |
| $Torque Capability$ | Nm | Torque capability beyond the minimum requirements | ACS |
| $Momentum Storage$ | Nms | Momentum storage beyond the minimum requirements | ACS |
| $Power Requirement$ | W | Peak power required by the ACS system | ACS |
| $Mass$ | kg | Total mass of the subsystem | ESS PMAD EGS ACS |
| $Volume$ | m ³ | Total volume of the subsystem | ESS PMAD EGS ACS |
| $Recurring Cost$ | \$k | Cost to produce one unit of the subsystem | ESS PMAD EGS ACS |
| $Non-recurring Cost$ | \$k | Cost to research and develop the subsystem | ESS PMAD EGS ACS |
| $Reliability$ | | Reliability of the subsystem | ESS PMAD EGS ACS |
| $ACS Eclipse Energy$ | Whr | Energy requirement to operate ACS during eclipse | ACS |
| E_{ACS} | Whr | Energy requirement to operate ACS over a single orbit | ACS |
| $ACS Flywheels$ | | Minimum number of flywheels necessary for ACS | ACS |

2.2.3.5 Flywheel Energy Storage. The purpose of the flywheel energy storage model is to determine the energy storage capacity, housekeeping power, mass, volume, reliability, recurring cost, and non-recurring cost of a flywheel energy storage unit.

Today there are no flywheel designs available for use on satellites. Information on designs is limited because the technology is still under development. Present flywheels are prototypes and probably are significantly different in detail from those that will be built in the future. Enough information does exist to model the performance of flywheels if the assumptions listed in 2.2.3.1 are made.

Equation 2.1 is used to calculate the energy storage capacity of the flywheel using the rotor inertia and the minimum and maximum rotational speed.

$$FlyCapacity = \frac{1}{2 \times 3600} I_s (\omega_{max}^2 - \omega_{min}^2) \quad (2.1)$$

The flywheel housekeeping power, the electrical load the flywheel imposes upon the satellite, is found by adding together the magnetic bearing and controller power requirements.

$$FlyPower = ControllerPower + BearingPower \quad (2.2)$$

Flywheel mass and volume are not computed. The values stored in the database are reported along with the other calculated outputs.

The flywheel model computes default non-recurring and recurring costs for each flywheel using cost estimating relationships (CER) taken from Space and Missile Systems Center's *Unmanned Space Vehicle Cost Model* (25). The formula uses the Energy Storage Suite, component-level, minimum unbiased percentage error CERs. These CERs provide cost estimates in thousands of fiscal year 1992 dollars (FY92 \$k). The model allows for the CER estimates to be adjusted to a different baseline year. The dollar multiplier adjusts costs to the new baseline year.

The non-recurring costs are estimated using the mass of one flywheel. The equation for computing the non-recurring costs for a flywheel not rated for launch is shown below

in Equation 2.3. The non-recurring cost for a flywheel rated for operation during launch is computed with Equation 2.4.

$$FlyNonRecurringCost = (DolMult)(NRCF)(14.523 \times FlyMass \times 2.2) \quad (2.3)$$

$$FlyNonRecurringCost_{LR} = (DolMult)(NRCF) \\ (14.523 \times [(FlyMass + FlyMass_{LR}) \times 2.2]) \quad (2.4)$$

Additionally, the model adjusts non-recurring cost estimates by a development factor taken from Larson (17). This factor is called the non-recurring cost factor (NRCF). Its value depends on the estimated level of research, development, test, and evaluation (RDT&E) required to produce space-qualified flywheel designs. Values greater than one increase the RDT&E estimates, while values less than 1 decrease the estimate. The NRCF estimated for flywheels is 1.5 to account for the lack of experience with, and limited knowledge of, composite flywheel technology.

Recurring, or per unit costs, are also computed using the mass of the flywheel. The equations for computing the recurring costs for each flywheel are shown below in Equation 2.5 and Equation 2.6 for launch-rated flywheels.

$$FlyCost = (DolMult)(7.939)(FlyMass \times 2.2)^{0.965} \quad (2.5)$$

$$FlyCost_{LR} = (DolMult)(7.939)[(FlyMass + FlyMass_{LR}) \times 2.2]^{0.965} \quad (2.6)$$

2.2.3.6 Battery Energy Storage. The purpose of the battery energy storage model is to determine the energy storage capacity, mass, volume, reliability, recurring cost, and non-recurring cost of a battery energy storage unit.

In many space applications, the battery chosen for a particular satellite mission is not designed specifically for that mission. Often, engineers choose an off-the-shelf battery for their satellite because that particular battery design has proven performance in the space environment (26). The spacecraft designers then simply choose the proper number of batteries to meet the energy and design life requirements of the mission.

The system-level model uses the above approach to determine the number of batteries required for the mission under evaluation. First, it computes the performance parameters for the particular chemical battery under evaluation based on mission requirements. The system-level model selects the appropriate number of batteries required to meet the mission's energy storage and power requirements.

The model computes the effective energy storage capacity (*Effective Capacity*) of each battery under consideration. The energy available from a battery with a given rated capacity in amp-hours is the product of the rated capacity, the bus voltage and the recommended depth-of-discharge (DOD). This is a simplifying assumption. The battery voltage will vary throughout the charge/discharge cycle and is typically not equal to the bus voltage. *Effective Capacity* is calculated using Equation 2.7.

$$\begin{aligned} \text{Effective Capacity (Whr)} &= \text{Bus Voltage} \times \text{Rated Capacity} \\ &\times \text{Recommended DOD} \end{aligned} \quad (2.7)$$

The battery hardware model computes the *Recommended DOD* based on the type of chemical battery (Ni-Cd, Ni-H₂, or Li-Ion) and the number of charge/discharge cycles (CDC) predicted for the mission orbit. For Li-Ion batteries, cycle life does not depend on depth-of-discharge. Therefore, the *Recommended DOD* is 100%. Current Li-Ion technology reports charge/discharge cycle lives of 1500-2000 cycles for Li-Ion batteries, therefore the model is only applicable with Li-Ion batteries for missions requiring less than 2000 CDCs (4).

For Nickel-Cadmium batteries, the *Recommended DOD* is found using Equation 2.8 (8).

$$\text{Recommended DOD} = -0.314 \log(\text{Total CDC Required}) + 1.562 \quad (2.8)$$

For Nickel-Hydrogen batteries, the *Recommended DOD* is found using Equation 2.9. (8)

$$\text{Recommended DOD} = -0.300 \log(\text{Total CDC Required}) + 1.785 \quad (2.9)$$

Total CDC Required is the total number of CDCs the model predicts the mission will encounter over the desired mission life.

To compute *Total CDC Required*, the model breaks the orbital altitude range into three regions: low-Earth orbits with altitudes ranging from 200 - 2,500 km, a broad mid-Earth orbit (MEO) region ranging from 2,500 - 35,786 km, and a geosynchronous region at the altitude of 35,786 km. For each of the three altitude regions, the model first computes the predicted number of charge/discharge cycles per year, and then multiplies this number by the *Design Life* to compute *Total CDC Required*.

The model assumes that the spacecraft encounters a charge/discharge cycle each orbit for LEO missions. The spreadsheet computes the number of charge/discharge cycles per year for low-Earth orbits based on the integer number of orbits per day multiplied by 365 days per year.

The model computes the number of CDCs per year for mid-Earth orbits by assuming there are two eclipse seasons per year and each eclipse season averages 50 days in duration (12). It then multiplies the number of orbits per day by 100 eclipse days per year to compute the CDCs per year.

There are 90 charge/discharge cycles per year for geosynchronous orbits. This number is based on one charge/discharge cycle per orbit, one orbit per day, and two 45-day eclipse seasons per year for geosynchronous orbits.

Battery mass and volume are not computed. The values stored in the database are reported along with the other calculated outputs.

The model computes default non-recurring and recurring costs for each battery using CERs taken from Space and Missile Systems Center's *Unmanned Space Vehicle Cost Model* (25). The equation uses the Energy Storage Suite, component-level, minimum unbiased percentage error CERs. These CERs provide non-recurring and recurring cost estimates in FY92 \$. The model allows for the cost estimates to be adjusted to a different baseline year. The dollar multiplier adjusts the costs to the new baseline year.

The non-recurring costs are estimated using the mass of one battery and its rated capacity. The equation for computing the non-recurring costs is shown in Equation 2.10.

$$\begin{aligned} \text{Non-recurring Cost} = & (\text{DolMult} \times \text{NRCF})[115.368 + (2.240 \\ & \times \text{Rated Capacity} \times \text{Battery Mass} \times 2.2)] \quad (2.10) \end{aligned}$$

Similar to the flywheel CERs, the model adjusts non-recurring costs by a development factor which modifies the non-recurring development cost estimates (17). This factor is called the non-recurring cost factor (NRCF). Its value depends on the estimated level of research, development, test, and evaluation (RDT&E) required for a particular battery design. The NRCFs for the three battery types considered are listed below:

1. Ni-Cd = 0.5
2. Ni-H₂ = 0.5
3. Li-Ion = 1.5

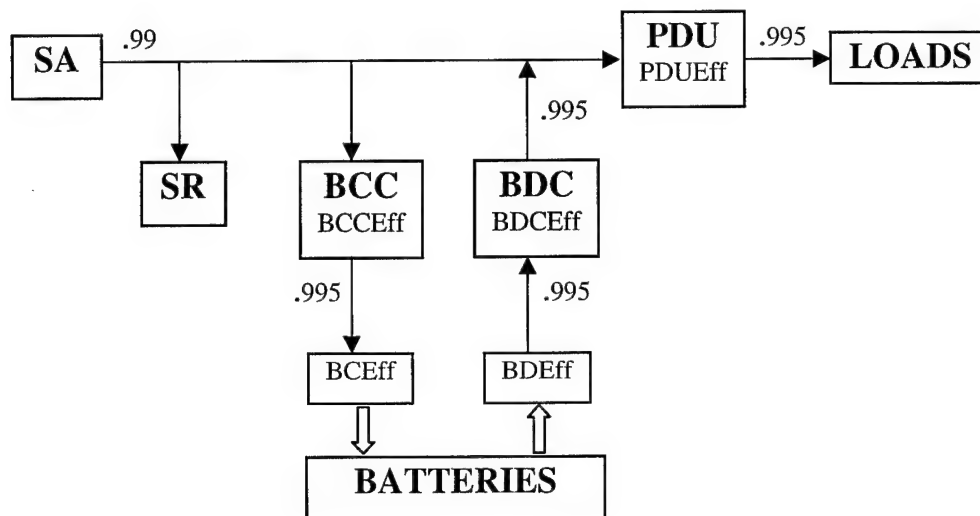
The model uses 0.5 for Ni-Cd and Ni-H₂ NRCFs to reflect the off-the-shelf nature of the pre-specified Ni-Cd and Ni-H₂ batteries in the database. These battery technologies are mature, and a full scale RDT&E program would not be necessary for these batteries. The model uses 1.5 as the NRCF for the Li-Ion batteries in the database. Li-Ion battery technology is relatively new, and the high NRCF captures this fact by increasing the non-recurring cost estimates above that given by the CER.

Recurring, or per unit costs are computed using the mass of the battery. The equation for computing the recurring costs for each battery is shown in Equation 2.11.

$$\text{Recurring Cost} = (\text{DolMult})(7.939)(\text{Battery Mass} \times 2.2)^{0.965} \quad (2.11)$$

2.2.3.7 Power Management and Distribution. One objective of this model is to account for the inefficiencies of the PMAD components. Power is passed between the solar arrays, the ESS, and the loads through the PMAD system. There are losses associated with the flow of power through these components. These losses must be

Battery Energy Storage DET



SA - Solar Array

SR - Shunt Regulator

PDU - Power Distribution Unit

BCC - Battery Charge Controller

BDC - Battery Discharge Controller

Eff - Efficiency

Figure 2.1 DET PMAD System - Battery ESS

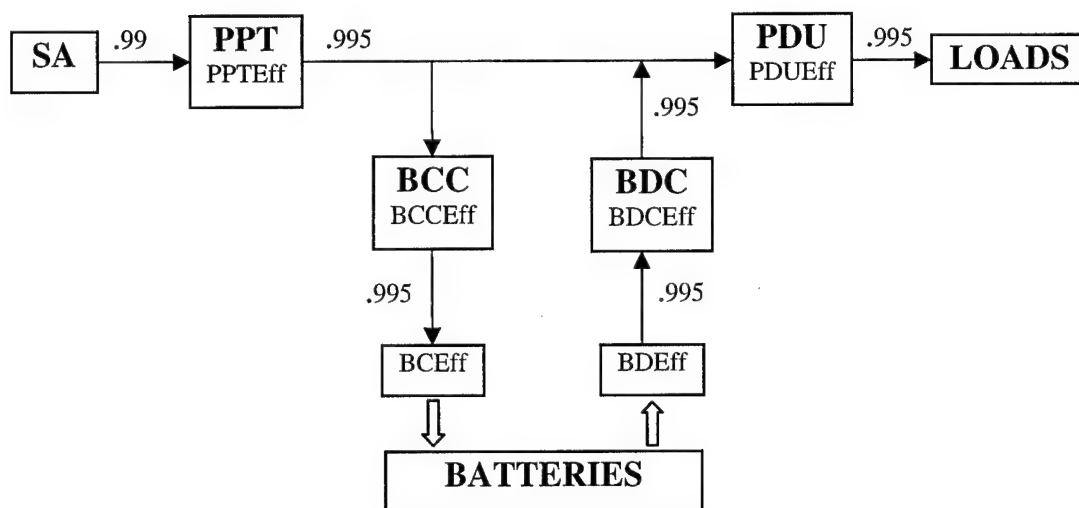
considered when sizing the solar arrays and ESS. Another objective is to provide estimates of the relevant PMAD system characteristics. This is done for the PMAD architecture with a battery ESS and for the PMAD architecture with a flywheel ESS. The characteristics under consideration are mass, cost, volume, and reliability.

The model is based on two fundamental architectures representing typical configurations. The first is a DET system, with the shunt regulator in parallel with the solar arrays. Thus, the shunt regulator causes no losses. The second architecture uses a PPT, non-dissipative means of regulation. A PPT introduces inefficiencies due to its presence.

The model's approach is to trace the route of power through the system. The PMAD hardware model accounts for efficiencies and calculates required outputs.

Figures 2.1, 2.2, 2.3, and 2.4 illustrate the architectures used in the PMAD model.

Battery Energy Storage PPT



SA - Solar Array

PPT - Peak Power Tracker

PDU - Power Distribution Unit

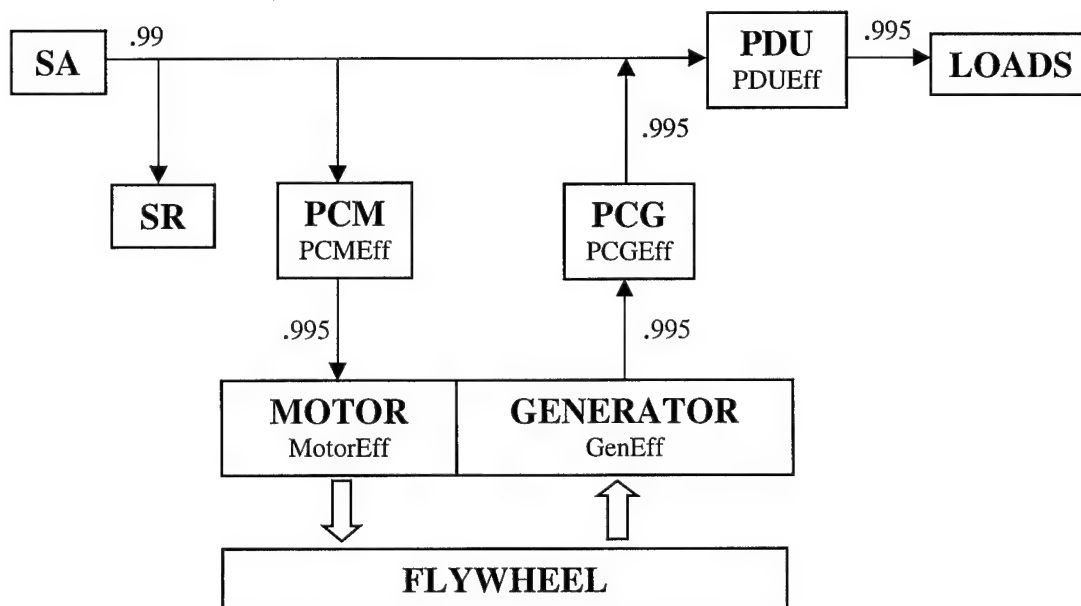
BCC - Battery Charge Controller

BDC - Battery Discharge Controller

Eff - Efficiency

Figure 2.2 PPT PMAD System - Battery ESS

Flywheel Energy Storage



SA - Solar Array

SR - Shunt Regulator

PDU - Power Distribution Unit

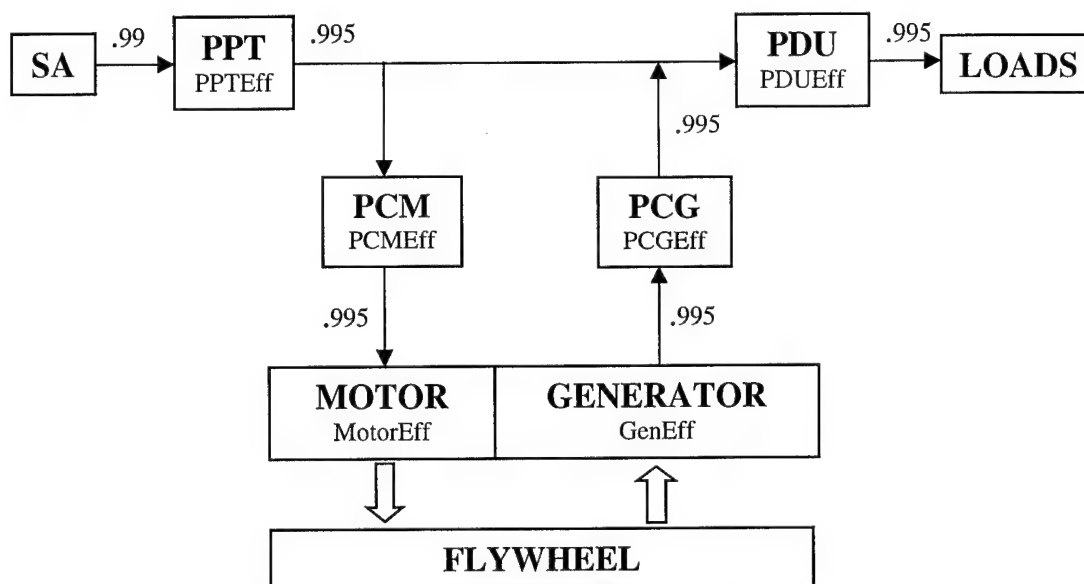
PCM - Power Converter for Motoring

PCG - Power Converter for Generating

Eff - Efficiency

Figure 2.3 DET PMAD System - Flywheel ESS

Flywheel Energy Storage



| | |
|-------------------------------|--------------------------------------|
| SA - Solar Array | PCM - Power Converter for Motoring |
| PPT - Peak Power Tracker | PCG - Power Converter for Generating |
| PDU - Power Distribution Unit | Eff - Efficiency |

Figure 2.4 PPT PMAD System - Flywheel ESS

For a battery ESS and a flywheel ESS, a certain amount of energy, E_{Req} , must be stored while the satellite is in sunlight. Also, the energy generation model must be given the solar array output power requirement. For a battery ESS, this is calculated in two parts. First, there is the portion of solar array output power required to store energy in the batteries at EOL, $P_{ES,EOL}$, and second, the portion required to support the loads, P_{Op} . When dealing with a flywheel ESS, $P_{EG,EOL}$ is the total solar array output power at EOL. The PMAD system efficiencies and satellite power load profile are used to calculate this quantity.

The attitude control model uses three PMAD efficiencies in order to produce required outputs. These are η_{Fly} , which is the total efficiency for power transfer between the flywheels; η_{EG-ES} , the total efficiency for power transfer from the solar array output to the flywheel input; and η_{EG-L} , the total efficiency for power transfer from the solar array output to the load input.

The final outputs are the required PMAD characteristics that will form part of the objective measures. This includes, M_{PMAD} , which is the mass of the PMAD system, $C_{PMAD,NR}$, the non-recurring cost of the PMAD system, and, $C_{PMAD,R}$, the recurring cost of the PMAD system. Also, there is the volume of the PMAD system, V_{PMAD} , and the reliability of the PMAD system, R_{PMAD} .

The following equations produce the outputs for a PMAD system used in conjunction with a battery ESS.

- Energy Storage Requirement

$$E_{Req} = \frac{E_{MaxEcl}}{\eta_{Batt-L}} \quad (2.12)$$

where

$$\eta_{Batt-L} = 0.995 * \eta_{PDU} * 0.995 * \eta_{BDC} * 0.995 * \eta_{BD} \quad (2.13)$$

The expression η_{Batt-L} represents the total efficiency from the battery output to the load input.

- EOL Power for Energy Storage

$$P_{ES,EOL} = \frac{P_{Batt,EOL}}{\eta_{EG-Batt}} \quad (2.14)$$

where

$$P_{Batt,EOL} = \frac{E_{Req}}{60 * T_{Charge,EOL}} \quad (2.15)$$

$$T_{Charge,EOL} = \begin{cases} 15 * T_{MaxEcl} & \text{if } T_{Min} > 15 * T_{MaxEcl} \\ T_{Min} & \text{otherwise} \end{cases} \quad (2.16)$$

$$T_{Min} = T_{Orb} - T_{MaxEcl} \quad (2.17)$$

and

$$\eta_{EG-Batt} = \begin{cases} \eta_{BC} * 0.995 * \eta_{BCC} * 0.99 & \text{if DET} \\ \eta_{BC} * 0.995 * \eta_{BCC} * 0.995 * \eta_{PPT} * 0.99 & \text{if PPT} \end{cases} \quad (2.18)$$

The expression $P_{Batt,EOL}$ represents the power requirement at the battery input for energy storage at EOL. $T_{Charge,EOL}$ is the time available to recharge the batteries at EOL. The expression T_{Min} is the minimum period that the satellite will be in sunlight. $\eta_{EG-Batt}$ represents the total efficiency from the solar array output to the battery input.

- Satellite Operating Power

$$P_{Op} = \frac{P_1 + P_{ACS}}{\eta_{EG-L}} \quad (2.19)$$

where

$$\eta_{EG-L} = \begin{cases} 0.995 * \eta_{PDU} * 0.99 & \text{if DET} \\ 0.995 * \eta_{PDU} * 0.995 * \eta_{PPT} * 0.99 & \text{if PPT} \end{cases} \quad (2.20)$$

The expression η_{EG-L} represents the total efficiency from the solar array output to the load input.

- PMAD Mass

$$M_{PMAD} = \begin{cases} M_{SR} + M_{BCC} + M_{BDC} + M_{PDU} + M_{Wire} & \text{if DET} \\ M_{PPT} + M_{BCC} + M_{BDC} + M_{PDU} + M_{Wire} & \text{if PPT} \end{cases} \quad (2.21)$$

where

$$M_{SR} = \begin{cases} 0.025 * P_{ES,BOL} * 0.99 & \text{if DET} \\ 0 & \text{if PPT} \end{cases} \quad (2.22)$$

$$M_{PPT} = \begin{cases} 0 & \text{if DET} \\ 0.025 * (P_{ES,BOL} + P_{Op}) * 0.99 & \text{if PPT} \end{cases} \quad (2.23)$$

$$M_{BCC} = \begin{cases} 0.025 * P_{ES,BOL} * 0.99 & \text{if DET} \\ 0.025 * P_{ES,BOL} * 0.99 * \eta_{PPT} * 0.995 & \text{if PPT} \end{cases} \quad (2.24)$$

$$M_{BDC} = 0.025 * \frac{P_1}{0.995 * \eta_{PDU} * 0.995 * \eta_{BDC}} \quad (2.25)$$

$$M_{PDU} = 0.02 * \frac{P_1}{0.995 * \eta_{PDU}} \quad (2.26)$$

$$M_{Wire} = 0.025 * M_{Dry} \quad (2.27)$$

The expressions M_{SR} , M_{PPT} , M_{BCC} , M_{BDC} , M_{PDU} , and M_{Wire} represent the mass of the Shunt Regulator (SR), PPT, BCC, BDC, PDU, and wiring harnesses respectively.

- PMAD Cost

$$C_{PMAD,NR} = 416.033 + 23.754 * 2.203 * M_{PMAD} \quad (2.28)$$

and

$$C_{PMAD,R} = 55.984 * 2.203 * M_{PMAD}^{0.698} \quad (2.29)$$

The expressions $C_{PMAD,NR}$ and $C_{PMAD,R}$ represent the non-recurring cost and the recurring cost of the PMAD system, respectively. Costs are based on CERs from the *Unmanned Space Vehicle Cost Model* (25)

- PMAD Volume

$$V_{PMAD} = 0.01 + 0.01 + 2 * 0.0008 * N_{Batt} \quad (2.30)$$

The first two terms represent the volume of the shunt regulator, or PPT, and the PDU. The last term represents the volume of the BCCs and BDCs given N_{Batt} .

The following equations produce the outputs for a PMAD system used in conjunction with a flywheel ESS.

- Energy Storage Requirement

$$E_{Req} = \frac{E_{MaxEcl}}{\eta_{Fly-L}} \quad (2.31)$$

where

$$\eta_{Fly-L} = 0.995 * \eta_{PDU} * 0.995 * \eta_{PCG} * 0.995 * \eta_{Gen} \quad (2.32)$$

The expression η_{Fly-L} represents the total efficiency from the flywheel output to the load input.

- Energy Generation Output Power Requirement at EOL

$$P_{EG,EOL} = \frac{P_{Fly}}{\eta_{EG-Fly-L}} \quad (2.33)$$

where

$$P_{Fly} = \frac{\sum_{i=1}^5 P_i * T_i + E_{GEO} * 60 + E_{Slew} * 60 + P_{Wheel} * T_{Orb}}{T_{Min}} \quad (2.34)$$

and

$$\eta_{EG-Fly-L} = \begin{cases} 0.995 * \eta_{PDU} * 0.995 * \eta_{PCG} * 0.995 * \eta_{Gen} * \\ \quad \eta_{Mot} * 0.995 * \eta_{PCM} * 0.99 & \text{if DET} \\ 0.995 * \eta_{PDU} * 0.995 * \eta_{PCG} * 0.995 * \eta_{Gen} * \\ \quad \eta_{Mot} * 0.995 * \eta_{PCM} * 0.995 * \eta_{PPT} * 0.99 & \text{if PPT} \end{cases} \quad (2.35)$$

P_{Fly} is the load and flywheel input power required to provide energy to the loads and store E_{MaxEcl} during T_{Min} . The expression $\eta_{EG-Fly-L}$ represents the total efficiency from the solar array output through the flywheels to the load input.

- Total Efficiency for Power Transfer Between Wheels

$$\eta_{Fly} = \eta_{Gen} * 0.995 * \eta_{PCG} * 0.995 * \eta_{PCM} * 0.995 * \eta_{Mot} \quad (2.36)$$

- Total Efficiency from Energy Generation Output to Flywheel Input

$$\eta_{EG-ES} = \begin{cases} 0.99 * \eta_{PCM} * 0.995 * \eta_{Mot} & \text{if DET} \\ 0.99 * \eta_{PPT} * 0.995 * \eta_{PCM} * 0.995 * \eta_{Mot} & \text{if PPT} \end{cases} \quad (2.37)$$

- Total Efficiency from Energy Generation Output to Load Input

$$\eta_{EG-L} = \begin{cases} 0.99 * \eta_{PDU} * 0.995 & \text{if DET} \\ 0.99 * \eta_{PPT} * 0.995 * \eta_{PDU} * 0.995 & \text{if PPT} \end{cases} \quad (2.38)$$

- PMAD Mass

$$M_{PMAD} = \begin{cases} M_{SR} + M_{PCM} + M_{PCG} + M_{PDU} + M_{Wire} & \text{if DET} \\ M_{PPT} + M_{PCM} + M_{PCG} + M_{PDU} + M_{Wire} & \text{if PPT} \end{cases} \quad (2.39)$$

where

$$M_{SR} = \begin{cases} 0.025 * P_{ES,Fly} * 0.99 & \text{if DET} \\ 0 & \text{if PPT} \end{cases} \quad (2.40)$$

$$M_{PPT} = \begin{cases} 0 & \text{if DET} \\ 0.025 * P_{EG,BOL} * 0.99 & \text{if PPT} \end{cases} \quad (2.41)$$

$$M_{PCM} = \begin{cases} 0.025 * P_{ES, Fly} * 0.99 & \text{if DET} \\ 0.025 * P_{ES, Fly} * 0.99 * \eta_{PPT} * 0.995 & \text{if PPT} \end{cases} \quad (2.42)$$

$$M_{PCG} = 0.025 * \frac{P_1}{0.995 * \eta_{PDU} * 0.995 * \eta_{PCG}} \quad (2.43)$$

where

$$P_{ES, Fly} = P_{EG, BOL} - \frac{P_{Min}}{\eta_{EG-L}} \quad (2.44)$$

$P_{ES, Fly}$ is the maximum power from the solar array output that can be directed to the flywheels. This energy can be stored in the flywheels, or it can be shunted. The equations for M_{PDU} and M_{Wire} are the same as for a PMAD system with a battery ESS. The expressions M_{PCM} and M_{PCG} represent the mass estimates for the PCM and the PCG respectively.

- PMAD Cost

$$C_{PMAD, NR} = 416.033 + 23.754 * 2.203 * M_{PMAD} \quad (2.45)$$

and

$$C_{PMAD, R} = 55.984 * 2.203 * M_{PMAD}^{0.698} \quad (2.46)$$

The expressions $C_{PMAD, NR}$ and $C_{PMAD, R}$ have the same interpretation as for a PMAD system with a battery ESS. Once again, costs are based on CERs from the *Unmanned Space Vehicle Cost Model* (25)

- PMAD Volume

$$V_{PMAD} = 0.01 + 0.01 + 2 * 0.0008 * N_{Fly} \quad (2.47)$$

The first two terms represent the volume of the shunt regulator, or PPT, and the PDU. The last term represents the volume of the PCMs and PCGs given N_{Fly} .

2.2.3.8 Energy Generation. The purpose of the energy generation model is to determine the mass, volume, reliability, recurring cost, and non-recurring cost of the energy generation system.

The PMAD model computes power requirements the energy generation system must satisfy. This power level must be generated by the solar array while in sunlight at end-of-life.

Given a battery energy storage system, this power requirement, P_{wrThru} , is the sum of P_{OpBatt} , the operating power PMAD needs from EG for loads at EOL, and $P_{RpEOLBatt}$, the recharge power PMAD needs from EG to recharge batteries at EOL.

Given a FES system, the power requirement, P_{wrThru} , is $P_{TotEOLFly}$, the total power PMAD needs from EG at EOL.

- Solar Array Area

The mounting coefficient, $MountingCoef$, is the ratio of the actual area of the solar array to its effective area, the area effectively receiving direct sunlight.

- For a panel-mounted, planar array, $MountingCoef = 1$.
- For a body-mounted, cylindrical array, $MountingCoef = 2.5$. This is lower than would be expected given the geometry of cylindrical arrays. Performance increases when the solar array cools as it spins into and out of sunlight. (16)
- For a body-mounted, omni-directional (spherical) array, $MountingCoef = 4$.

The temperature of a solar array is a function of altitude. It typically ranges from 67 deg C in LEO to 53 deg C in GEO (16). Assuming a linear interpolation,

$$ArrayTemp \text{ (deg C)} = (-0.0004)(Altitude) + 67.3 + 5 \text{ (if body-mounted)} \quad (2.48)$$

The temperature inefficiency of the solar array material is η_{TempSA} . This is the reduction in array performance experienced for every degree Celsius the temperature of the array exceeds 28 deg C (16). The temperature inefficiency of the array is

$$TempIEff = \eta_{TempSA}(ArrayTemp - 28) \quad (2.49)$$

The efficiency of the solar array material is $ArrayEff$. Traditional values range from 11% to 18% for silicon and gallium arsenide, respectively. The solar array must

be sized to meet power requirements when solar radiation intensity is at a minimum. This occurs on July 4th, when Earth is farthest from the Sun. The minimal solar intensity experienced, *MinSInt*, is $1,309 \text{ W/m}^2$ (2). Therefore, the minimal areal power density of the array, the smallest power produced per unit area if sunlight strikes all of the solar cells, is

$$\text{ArealPwrDen } (W/m^2) = 1309 \text{ W/m}^2 \times \text{ArrayEff} \quad (2.50)$$

The efficiency of the solar array related to design, assembly, shadowing, and optical losses of the array is η_{DesSA} . The overall inefficiency of the solar array is

$$\text{TotalInEff} = 1 - [\eta_{DesSA}(1 - \text{TempIEff})] \quad (2.51)$$

The maximum angle of incidence of sunlight on the solar array is *MaxAngInc*. The areal power density of the array at beginning-of-life (BOL), with efficiencies and incidence incorporated, is

$$\begin{aligned} \text{ArealPwrDenBOL } (W/m^2) &= \text{ArealPwrDen} \times (1 - \text{TotalInEff}) \\ &\times \cos(\text{MaxAngInc}) \end{aligned} \quad (2.52)$$

The mission design life of the satellite is *Design Life*, and the annual degradation of the array due to meteors and solar radiation is *AnnDegSA*. The degradation of the array becomes

$$\text{LifeDegrad} = (1 - \text{AnnDegSA})^{\text{Design Life}} \quad (2.53)$$

and the areal power density at end-of-life (EOL) becomes

$$\text{ArealPwrDenEOL } (W/m^2) = \text{ArealPwrDenBOL} \times \text{LifeDegrad} \quad (2.54)$$

The area of the solar array, *ArrayArea*, can then be computed using the power PMAD requires from the energy generation system, *PwrThru*.

$$ArrayArea (m^2) = MountingCoeef \frac{PwrThru}{ArealPwrDenEOL} \quad (2.55)$$

- Solar Array Volume

According to Agrawal (2), typical diameters of solar array booms range from 2.2 to 3.4 cm. Averaging these values and adding 1.3 cm of thickness for the array and the coverglass, we estimate the total thickness of the array to be approximately 4 cm. This value also works for body-mounted arrays, since the mounting would use space not used for the booms. The volume of the array can then be computed as

$$ArrayVol (m^3) = 0.04 \times ArrayArea \quad (2.56)$$

- Solar Array Mass

The specific power of the solar array material is *ArraySP*. The specific power of the array at BOL can be computed as follows:

$$SASpPwrBOL (W/kg) = ArraySP \times (1 - TotalInEff) \times \cos(MaxAngInc) \quad (2.57)$$

Specific power at EOL, *SASpPwrEOL*, takes into account the degradation over the array's lifetime.

$$SASpPwrEOL (W/kg) = SASpPwrBOL \times LifeDegrad \quad (2.58)$$

The solar array's mass is then

$$ArrayMass (kg) = \frac{PwrThru}{SASpPwrEOL} \quad (2.59)$$

- Solar Array Reliability

Our research did not uncover any information regarding the probability a solar array will still be performing its function at the satellite's end-of-life. It appears to be generally assumed that a catastrophic failure is very unlikely. For these reasons, the energy generation model reports a solar array reliability, $ArrayRel = 0.999$.

- Solar Array Recurring and Non-recurring Costs

Cost estimating relationships for solar arrays were determined by Air Force Materiel Command's Space and Missile Systems Center (25). These CERs estimate the recurring and non-recurring costs for a solar array as a function of the array's mass, power output at beginning-of-life, and number of solar cells. The size of the solar cells was assumed to be 4 cm x 6 cm.

The number of solar cells, $NumSCells$, is

$$NumSCells = \frac{ArrayArea}{0.0024 \text{ m}^2} \quad (2.60)$$

According to Space and Missile Systems Center, the recurring costs are

$$ArrayCost \text{ (FY92 \$k)} = 4.959 \times NumSCells^{0.621} \quad (2.61)$$

The non-recurring costs are

$$\begin{aligned} ArrayNRCost \text{ (FY92 \$k)} = & 0.024 \times ArrayMass \times \frac{PwrThru}{LifeDegrad} \\ & + 0.021 \times NumSCells \end{aligned} \quad (2.62)$$

2.2.3.9 Attitude Control Model. The objective of the attitude control model is to provide results for the seven performance measures and to calculate power and energy requirements. The data is then passed to the other models. The performance measures for the attitude control model are mass, non-recurring cost, recurring cost, volume, reliability, torque capability, and momentum storage capacity. These performance measures evaluated the different hardware combinations for their performance in attitude

control. The following is a description of the ACS power requirement and the performance measures.

- Power Requirement

Power budgets are an important part of spacecraft. The spacecraft is limited to the energy stored by the ESS or to the energy provided by the solar arrays. In general, it is important to minimize the power requirements of subsystems. The three ACS methods had different power requirements and different ways of measuring those power requirements. Energy requirements are used in determining the impact of the different ACS methods on the number of energy storage units required.

MWA This method stores energy without affecting attitude, assuming perfect control of the flywheel is possible. In real systems, energy storage would cause small torques due to imperfect control. Power requirements are the efficiency losses as energy is moved from one flywheel to another to maintain attitude. During energy storage and energy removal, these losses due to attitude control may not occur if all the torques to store or remove energy are greater than the required attitude torques. There are still losses due to storing or removing energy, but we account for these in a separate part of the SLM.

Hall shows that the minimum number of wheels necessary to store energy and control attitude using the MWA is 4 wheels (10). With N wheels, where N is the number of wheels, there are 3 torque modes in attitude control which also affect energy storage, and $N-3$ torque modes in energy storage which do not affect attitude control. These energy storage modes are the null space of the flywheel spin axes. The torques for controlling attitude changes the amount of energy stored in the flywheels. Given an amount of energy to be stored or removed from the flywheels, the remainder is stored or removed using the energy storage modes. A more detailed description is available in Appendix B.

By changing wheel speeds in multiples of a null space, the angular momentum of the spacecraft will not change. With only four wheels, there is only one energy mode. Therefore, any other combination of changes in wheel speeds cause a

change in the angular momentum and attitude of the satellite. A flywheel system can take advantage of this fact. The torques on the wheel decompose into a linear combination of the attitude control torque and the energy storage torque. As long as four non-coplanar wheels are operational, the flywheels can store energy and maintain attitude.

One benefit of the MWA is that if the torques associated with energy storage or removal are greater than the attitude control torques, there are no additional losses due to controlling attitude. However, throughout an orbit there may be cases where the attitude torques are greater than the energy storage torques, especially when the flywheels are nearly fully charged. This will result in energy being transferred from one wheel to another, rather than coming from the solar array or going to the loads. The energy requirement for attitude control is then the total energy transferred between wheels over an orbit multiplied by the interwheel inefficiency.

Calculating the worst-case energy requirement for the MWA is complicated. The process begins with the solar array and the orbit. At LEO, the eclipse period is a relatively large portion of the orbit. The spacecraft uses flywheels to provide power during eclipse. During the sunlit period, the solar array must provide enough power to cover loads plus enough power to recharge the flywheels. The attitude control requires the most power if the ACS torque's direction opposes, and its magnitude exceeds, the energy storage torques. To calculate the worst-case power requirements, the external torques are always assumed to oppose the energy torques. The greatest energy use for attitude control during eclipse occurs when loads are a minimum, and therefore energy torques are a minimum. A low energy torque maximizes the likelihood that energy is passed between wheels. This transfer is needed to maintain attitude. The low energy torque also means more energy is passed. However, the solar arrays in our model are designed to provide for maximum loads during eclipse. Since the power requirement for ACS is an efficiency loss, the change in power from minimum to maximum load is greater than the decrease in ACS power. Thus, the maximum

total eclipse energy coincides with maximum loads during eclipse. This is the case we consider in our model.

The power requirement for attitude control is the maximum efficiency losses for moving energy from one flywheel to another. The development of Equation 2.63 is long and is included in Appendix B.

$$P_{MWA} = (1 - \eta_{IW}) \left(\frac{N_{Fly} - X^2}{N_{Fly}} X T_{ext} - X^2 T_{en} \right) \omega \quad (2.63)$$

where X depends on the set of axes that caused the greatest energy transfer. Using X allowed a single equation to be developed.

$$X = \begin{cases} 0 & \text{if } \frac{N_{Fly}}{N_{Fly}-1} T_{en} > T_{ext} \\ 1 & \frac{N_{Fly}}{N_{Fly}-1} T_{en} < T_{ext} < \frac{N_{Fly}}{(\sqrt{2}-1)N_{Fly}+1-2\sqrt{2}} T_{en} \\ \sqrt{2} & \frac{N_{Fly}}{(\sqrt{2}-1)N_{Fly}+1-2\sqrt{2}} T_{en} < T_{ext} \end{cases} \quad (2.64)$$

The final set of equations calculate the maximum eclipse energy. The attitude energy requirement during a load period comes in two parts if X changes over the load period. The result is zero unless X is nonzero. When X is nonzero, the external torques are large enough to require energy transfers from one wheel to another during eclipse.

The development began by assuming a constant load on the flywheel, P_F . The attitude energy requirement is the integral of the power from Equation 2.63 from $t = t_{beg}$ to $t = t_{end}$, or $t = t_{ch}$, if it exists. The values of X and T_{ext} remain constant over the integral. Therefore, the integral of Equation 2.63 can be easily separated into two separate integrals. $T_{en}\omega$ is always P_F . The first integral requires more work. We know that

$$\omega(t) = \sqrt{\omega_{beg}^2 - \frac{P_F t}{I_s}} \quad (2.65)$$

Integrating results in

$$\int_0^t \frac{N_{Fly} - X^2}{N_{Fly}} X T_{ext} (1 - \eta_{IW}) \omega dt = \frac{2I_s(\omega_{beg}^3 - \omega_{end}^3)(1 - \eta_{IW}) \frac{N_{Fly} - X^2}{N_{Fly}} X T_{ext}}{3P_F} \quad (2.66)$$

Putting the two integrals together produces the attitude control energy requirement for a fixed X and P_F .

$$E_{ACS} = \sum_{load\ pds} \sum_X (1 - \eta_{IW}) \left(\frac{2(N_{Fly} - X^2) X T_{ext} I_s (\omega_{beg}^3 - \omega_{end}^3)}{3N_{Fly} P_F} - X^2 P_F LD \right) \quad (2.67)$$

The next energy requirement is due to any slew maneuvers. The maximum energy for a slew maneuver is

$$E_{ecl\ slew} = \sqrt{3} g_{slew} \omega_{max} (1 - \eta_{IW}) \min(t_{slew} N_{slew}, E_{cl\ max}) \quad (2.68)$$

The solar array must provide this additional ACS energy during the recharge period.

At GEO, the method of storing energy follows a different path. Rather than an eclipse every orbit, eclipse seasons include orbits during which no eclipse occurs. Also, eclipse durations never exceed 6% of the orbital period. The solar array provides sufficient power to cover the spacecraft's energy needs over the entire orbit. The flywheels provide the excess power needed during peak requirements in addition to powering the spacecraft during eclipses. This results in lower torques on the wheels due to energy storage, which increases the attitude control power usage. The largest power requirement occurs when the wheels are operating near maximum speed. We used the following formula to estimate the smallest torque necessary to perform energy storage.

$$T_{en} = \begin{cases} \frac{\eta_{EG-ES}(\eta_{EG-L}P_{SA}-P)}{\omega_{max}N_{Fly}} & \text{if } P < \eta_{EG-L}P_{SA} \\ \frac{\eta_{ES-L}(P-\eta_{EG-L}P_{SA})}{\omega_{max}N_{Fly}} & \text{otherwise} \end{cases} \quad (2.69)$$

Summing over the load periods gives the worst-case energy requirement for attitude control over an orbit. See Equation 2.63. Slew maneuvers accounted for additional energy according to Equation 2.68.

GFA This method required an additional load to power the gimbals, which must be moved to counter energy storage or removal in the wheels as well as external torques. However, GFA does not require energy transfer from one flywheel to another.

The gimbal power requirement is from Larson (16), where the steady state power per gimbal is 15 to 30 W in stand-by mode and an additional 0.02 to 0.2 W/Nm when torquing. The following formula provides a first estimate to modeling gimbaled flywheels using the magnetic bearings to provide the torque.

$$P_{gimbal} = 20\text{W standby} + 0.1\text{W/Nm when torquing} \quad (2.70)$$

TACA Power requirements are similar to the GFA listed above. The flywheel batteries are counter-rotated to cancel momentum and torque effects of storing and removing energy from the flywheels. Power requirements are derived from the off-the-shelf hardware information. For the ACS model, actual attitude control system data is used.

- **Torque Capability**

This output is a function of the hardware, external torques, stability requirements, slew rates, and flywheel energy storage torques. More torque capability for attitude control provides better stability and better slew rates. The minimum torque requirement is the maximum of

- The slew torque requirement or

- The sum of external torques, the torque necessary to maintain stability, and the torques due to flywheel energy storage.

The minimum torque requirement is assumed to be possible in any axis. The output torque capability is the torque capability of the ACS minus the minimum torque requirement.

All of the following torque equations are from Larson (16).

The equation for stability torque is derived for separation from the booster and assumes a tip-off rate which is the angular velocity at separation. The ACS requires a minimum torque to keep the spacecraft within the maximum attitude deviation. A maximum body spin rate is assumed, using a linear formula with respect to altitude. The outcome is a reasonable approximation of the necessary stability torques. Larger results, over 0.2 Nm, are much less accurate and should be examined more closely. The slew torque equation assumes full acceleration torque for half the maneuver and full deceleration torque for the remaining time. The external torques that must be overcome include gravity gradient, aerodynamic, magnetic, and solar pressure. The resulting external torque is the sum of the four external torques.

$$T_{stab} = \frac{\pi \omega_{maxbodyspinrate}^2 max(I_x, I_y, I_z)}{360 \theta_{maxdev}} \quad (2.71)$$

$$\omega_{maxbodyspinrate} = 0.0015 + (42164 - r)/4000000 rad/s \quad (2.72)$$

$$T_{sl} = \frac{4 max(I_x, I_y, I_z) \theta_{slew}}{t_{slew}^2} \quad (2.73)$$

$$T_{gg} = \frac{3\mu(I_z - min(I_x, I_y)) \sin 2\theta_{maxdev}}{2r^3} \quad (2.74)$$

$$T_{ggslew} = \frac{3\mu(I_z - min(I_x, I_y)) \sin 2\theta_{slew}}{2r^3} \quad (2.75)$$

$$T_{aero} = 0.5 \rho C_d A V^2 (c_{pa} - c_g) \quad (2.76)$$

$$T_{mag} = \frac{F_I D M}{r^3} \quad (2.77)$$

$$T_{solar} = \frac{K A (1 + q) (c_{ps} - c_g)}{c} \quad (2.78)$$

$$T_{ext} = T_{ggslew} + T_{aero} + T_{mag} + T_{solar} \quad (2.79)$$

MWA The ACS provide torques by changing flywheel speeds. Flywheels operate in roughly the same range as reaction wheels. It is the motor/generator that determines the limit of the torque possible. Torques for reaction wheels are typically from 0.03 to 2 Nm. These values are indicative of current reaction wheels and depend on the motor/generator. The MWA is assumed to be a zero momentum system, where all of the wheels are spinning at the same speed. Thus the total angular momentum of the flywheels is zero. This method requires a minimum of 4 wheels to provide both energy storage and attitude control. Four wheels are assumed to provide the torque capability of a single flywheel in each axis. To estimate the effect of additional wheels beyond the minimum of 4, extra wheels in groups of three provide the additional torque capability of a single wheel in each axis. Using less than an additional three wheels is estimated by assuming the wheel provides one-third of its torque capability in each primary axis. Since adding a wheel requires a change of orientation, this assumption is close but not exact.

$$T_{cap} = \frac{N_{Fly} - 1}{3} T_{mg} - \max[T_{stab} + T_{ext} + Res \max(T_{en}), T_{sl}] \quad (2.80)$$

GFA The ACS provides torques by changing the spin axes of the flywheel with respect to the spacecraft using a gimbal. Torques of control moment gyros (CMG) are typically 25 to 500 Nm. For flywheels, the angular momentum is higher, due to the high RPMs of flywheels in comparison to current CMGs. The increase in angular momentum allows a traditionally gimbaled flywheel to provide the same torque for a smaller gimbal deflection. Flywheels also have the option of magnetically gimbaling the flywheel. This method saves on mass, but makes the magnetic bearings more complicated. The equation approximates the gimbal torque output based on an assumed gimbal rate. The torque created by storing energy in the flywheels does not cancel in this method. The GFA needs to counter the full energy storage torque. This method requires a minimum of 2 wheels to provide both energy storage and attitude control. However, using only 2 wheels required a mixture of changing wheel speeds and spin axes. Therefore

for the GFA, the assumption is that wheel speeds are used strictly for energy storage, and changing the spin axes is the only method controlling attitude. Three wheels are assumed to provide the torque capability of a single flywheel in each axis. To estimate the effect of additional wheels beyond the minimum of 3, extra wheels in groups of three provide the additional torque capability of a single wheel in each axis. Using less than an additional three wheels is estimated by assuming the wheel provided a third of its torque capability in each primary axis.

$$T_{cap} = \frac{N_{Fly}}{3} I_s \omega_{max} \omega_{gimbal} - \max[T_{stab} + T_{ext} + \max(T_{en}), T_{sl}] \quad (2.81)$$

$$\omega_{gimbal} = 0.225 \text{ rad/s} \quad (2.82)$$

TACA The ACS provides torques either by changing reaction wheel speeds or by using CMGs. Torque ranges are the same as listed above, 0.02 Nm to 500 Nm, depending on the system used. We assume off-the-shelf hardware for TACA, and the torque capability is known. Flywheels are counter-rotated, and energy is removed at the same rate from both flywheels. Therefore, the net change in angular momentum is zero and there is no torque on the spacecraft. In actuality, there is a small torque due to energy storage which depends on how accurate the flywheel speeds are controlled.

$$T_{cap} = T_{taca} - \max[T_{stab} + T_{ext} + Res \max(T_{en}), T_{sl}] \quad (2.83)$$

- Momentum Storage

This value in the first estimate provides a rough measure of how well the spacecraft can handle cyclic external torques. It also provides a rough estimate on how frequently reducing excess momentum due to secular external torques is required.

$$\text{Maximum Momentum Storage} = \left\{ \begin{array}{ll} MWA & I_s(\omega_{max} - \omega_{min}) \\ GFA & I_s\omega_{max}\omega_{gimbalrate} \\ TACA & \text{hardware characteristic} \end{array} \right\} \quad (2.84)$$

The secular external torques depend on the orientation of the spacecraft. Larson (16) describes the relationship between orientation and type of torque, either secular or cyclic. Gravity gradient and aerodynamic forces produce secular torques, while solar radiation and magnetic torques are cyclic for Earth-pointing satellites. For an inertially pointing vehicle, solar radiation and aerodynamics forces produce secular torques, while gravity and magnetic torques are cyclic. The secular torques cause unwanted momentum storage that must eventually be reduced using magnetic torquers or thrusters. The cyclic torques produce the maximum disturbance in a quarter of an orbit. Excess momentum storage provides margin for larger than expected torques and buildup of secular torques.

These torques cause a problem for the momentum wheel approach. Momentum buildup in the MWA is directly linked with energy storage. The added momentum causes a single wheel to reach maximum speed before the other wheels, assuming the wheel speeds can not be raised above the maximum speed nor below the minimum speed. The ACS halts the addition of energy or momentum to this wheel, if possible. Since the other wheels have not reached their maximum speed, more energy can be stored in these wheels. However, if storing more energy would cause the single wheel to spin above its maximum speed, then the stored energy has reached a maximum. This results in unavailable energy storage. Likewise, a wheel may reach its minimum speed before all the other wheels.

Larson (16) develops the equations to calculate the maximum momentum a spacecraft will store due to external torques. Equation 2.94 calculates the worst-case energy storage capability decrease caused by excess momentum storage. The development assumes external torques are such that three of the flywheels did not store any momentum. If the minimum number of wheels cannot provide sufficient momentum

storage, additional wheels are added. These wheels are assumed for the purposes of calculations to be used strictly for attitude control and not for energy storage.

$$\text{cyclic} \quad \Delta H_{max} = \int g dt = g \frac{T_{Orb}}{4} (0.637) \quad (2.85)$$

$$\text{secular} \quad \Delta H_{max} = \int g dt = g T_{Orb} \quad (2.86)$$

$$\text{slews} \quad \Delta H_{max} = \int g dt = g_{slew} t_{slew} / 2 \quad (2.87)$$

$$\Delta H_{total} = \sum \Delta H_{max} \quad (2.88)$$

$$\Delta \omega = \frac{M_N \Delta H_{total}}{I} \quad (2.89)$$

$$\text{where} \quad M_N = \begin{cases} 1 & \text{if } N_{Fly} = 4 \\ \frac{\sqrt{2}}{2} & N_{Fly} = 5 \\ \frac{\sqrt{3}}{3} & N_{Fly} = 6 \\ 0.5 & N_{Fly} = 8 \end{cases} \quad (2.90)$$

$$E_{avail \ max} = N_{Fly} I (\omega_{max}^2 - \omega_{min}^2) / 2 \quad (2.91)$$

$$E_{avail \ min} = (N_{Fly} - 3) I (\omega_{max}^2 - \omega_{min}^2) / 2 + \frac{3}{2} I [(\omega_{max} - \Delta \omega)^2 - (\omega_{min} + \Delta \omega)^2] \quad (2.92)$$

$$\Delta E_{lost} = E_{avail \ max} - E_{avail \ min} \quad (2.93)$$

$$\Delta E_{lost} = 3 M_N \Delta H_{total} (\omega_{max} + \omega_{min}) \quad (2.94)$$

The dedicated ACS flywheels are added in groups of three. Therefore, each group reduces the lost energy in the other flywheels by reducing the momentum storage in energy storing wheels.

- Mass

Mass for the attitude control system is the mass of additional components required to perform attitude control. We assume attitude determination components would be the same regardless of the attitude control required. Also, we assume any impact on the computer system is identical for all alternatives and can be ignored.

MWA All the components of the attitude control are also required for energy storage; therefore the mass of attitude control is zero.

GFA The flywheels are also required for energy storage; mass of the flywheels counts against energy storage. Only the mass of the gimbaling mechanism counts towards the attitude control system. A gimbaling mechanism is required for each flywheel. The equation to estimate this value is

$$\text{GFA Mass per gimbal} = 3\text{kg} + 15\text{kg/m}^3 \times \text{FlyVolume} \quad (2.95)$$

TACA A complete attitude control system is required; hence the mass is that of the system. TACA mass is modeled using actual hardware specifications.

- Recurring Cost

The cost considered is the cost of additional components.

MWA No new components are necessary; the added cost is \$0.

GFA Added costs are derived from strengthening the magnetic bearings and adding either a tilt table or a gimbal device. The equation used to estimate this additional cost is roughly approximated with the following equation modified from Space and Missile Systems Center's CER (25):

$$\text{Cost per flywheel} = \$30000 (\text{Mass}_{\text{gimbal}} N_{\text{Fly}})^{0.62} \text{Life}^{0.473} \quad (2.96)$$

TACA TACA cost is the cost of the hardware being used.

- Non-recurring Cost

The cost considered is the cost of hardware RDT&E.

MWA This method has not been used in space yet. RDT&E costs are roughly approximated with the following equation modified from Space and Missile Systems Center's CER (25):

$$\text{Non-recurring Cost} = \$115000 (\text{Mass}_{\text{flywheel}} N_{\text{Fly}})^{0.5} \quad (2.97)$$

GFA Gimbale flywheels are still in development with on-going research into the magnetic bearings necessary to provide the requisite torques. The model uses the following modified CER from Space and Missile Systems Center (25) to roughly approximate the non-recurring costs to develop the technology and integrate into a spacecraft.

$$\text{Non-recurring Cost} = \$126000 \text{ } Mass_{gimbal}^{0.733} N_{Fly}^{0.5} \quad (2.98)$$

TACA TACA cost is the cost of the hardware being used.

- Volume

Volume is the additional volume required by attitude control.

MWA This method required no additional volume.

GFA This method requires additional volume beyond the flywheel volume. Volume is required not only for the gimbal mechanism but also to allow the flywheel to rotate. We chose to estimate this additional volume as the following:

Additional volume per flywheel = Original volume of the flywheel

TACA Volume is the volume of the hardware being used.

- Reliability

There are two components to reliability. The first is the effect flywheels have on reliability. The second is the effect of everything other than the flywheels. This section covers reliability of attitude control for the three methods. The only effects considered are those of flywheel failures and actuator failures. Failures in computers, attitude determination, thrusters, or magnetic torquers are not considered.

MWA Reliability for the MWA is the reliability of the flywheels. The minimum number of flywheels required to perform attitude control and simultaneous energy storage is 4. When the number of working flywheels drops to 3 or less, or the remaining flywheels are coplanar, the flywheels cannot perform both ACS and ESS functions simultaneously. As long as 4 non-coplanar flywheels are

working, and all non-functioning flywheels are decreased to zero speed, both attitude control and energy storage will be functional.

Possible effects of non-fatal loss of flywheels:

- Attitude control is no longer optimized
- Possible decrease in stability
- No longer using best sets of flywheels to provide specific torques
- Increase in power required to counter some torques
- Limited range of operation for maximum and minimum wheel speed
- Storing and removing energy from a flywheel uses a torque that must be countered by other flywheels. If a flywheel is lost, other flywheels must cover any torques it countered. If there are many more than 4 flywheels, this is not a problem. With a decrease from 5 to 4 flywheels, there is a large chance that at least one flywheel must reduce the range of speeds over which it operates.

GFA Reliability for the GFA consists of two parts. The first part is the reliability of the flywheels. The second part is the reliability of the actuators. Under the GFA, the flywheels are operated strictly for energy storage. The actuators provide the torque capability for attitude control. If a flywheel fails, the actuator is no longer useful because the flywheel speed is assumed to be zero. The actuator cannot provide torque using a stationary flywheel. One full flywheel is lost to attitude control. If the actuator fails, the assumption is that the actuator locks and can no longer provide torque. The flywheel can still be used for energy storage. Beginning with 3 flywheels with working actuators, if a flywheel fails the result is 2 flywheels with actuators. This is 4 degrees of freedom using the remaining flywheels in a differential rate mode similar to the MWA. Full attitude control would be possible. Energy storage would be severely limited or not possible at all. Beginning with 3 or more flywheels, as long as three working flywheel-actuator pairs are still working, full attitude control is possible at a lesser stability.

TACS Reliability for TACS is available for most commercially available systems.

Typical TACS use 4 flywheels, three orthogonal and one skewed. As long as three flywheels are working, full attitude control is possible.

2.2.4 Description of Systems. We modeled four system configurations.

These are listed below:

1. Traditional Attitude Control System with Battery Energy Storage (TACS/B)
2. Traditional Attitude Control System with Flywheel Energy Storage (TACS/FES)
3. Integrated Power and Attitude Control System/Momentum Wheel Approach (IPACS/MWA)
4. Integrated Power and Attitude Control System/Gimbaled Flywheel Approach (IPACS/GFA)

The details of each system are described below.

TACS include momentum wheels, reaction wheels, and control moment gyros. The energy storage system is modeled independently and comprises one or more batteries or pairs of flywheels.

IPACS/MWA use the same equipment (motor/generators, controllers, flywheels, and bearings) to provide energy storage and attitude control. The system size is the maximum of the minimum number of wheels required to meet attitude control requirements, energy storage requirements or reliability requirements. (equation 2.99).

$$n_{wheels} = \max\{n_{ESS}, n_{ACS}, n_{Reliability}\} \quad (2.99)$$

In an IPACS/MWA, the controller adjusts wheel speeds individually to change the satellite's angular momentum. The same wheel controller is also used to schedule energy drains to meet satellite power requirements.

IPACS/GFA use some common equipment (the flywheels and associated components) to provide angular momentum and energy storage, and have separate equipment (gimbals and actuator elements) for attitude control. In these systems, the number of wheels is determined using Equation 2.99. The controller adjusts wheel speed to meet power requirements, and the actuators change wheel orientation to meet satellite attitude control requirements.

2.2.5 System-Level Modeling. Once the functions were represented with equations, we collected the lower-level function models into a system-level model. The system-level model combines the appropriate functions necessary to form our defined system. The system-level model imposes system performance requirements and trades on the data computed by the lower level models. It is responsible for selecting, configuring, sizing, scoring and reporting the properties of a specific alternative design.

2.2.5.1 General Algorithm. The system-level model executes the following algorithm:

1. Pick a design alternative (a specific set of alterables)
2. Verify that the alternative is allowed
3. Verify that the alternative is desired
4. Configure the alternative as a system
5. Size the system to meet ACS requirements first, then to meet ESS, and finally to meet the reliability requirements, if necessary
6. Compute the system's performance
7. Compute utilities from the performance measures
8. Compute the objective measures from the utilities
9. Compute value score from the objective measures
10. Report the system characteristics and scores
11. Iterate until all alternatives have been examined

This section explains the details of the system-level model.

2.2.5.2 Alternative Selection and Configuration. The system-level model uses an exhaustive loop system to select alternative designs. There are four loops in the model, one for each alternative. They are:

ACS Scheme The attitude control scheme has eight possible values. The attitude control scheme is either IPACS/MWA, IPACS/GFA, or one of the six TACS in the database.

ESS Scheme The energy storage scheme has 12 possible values. These correspond to the six flywheel and six battery data sets in the database.

PMAD Scheme The power management and distribution scheme has two possible values. They identify whether the system is a peak power tracking or direct energy transfer design.

Launch Scheme The launch scheme has two possible values. The locked value indicates that the flywheels are not used until after the solar arrays are deployed. This system requires a primary power supply that it uses until the solar arrays are deployed and the flywheels are spun up. The spinning value indicates that the flywheels provide primary power during the launch-to-deployment phase of the mission. This option requires a robust design that can operate in the launch environment.

The model considers, at most, 384 alternatives. Of course some of the possible alternatives do not make any sense. For example, batteries combined with a spinning launch scheme is an unreasonable combination. Similarly, IPACS are incompatible with batteries. The configuration method in the system model is used to determine which systems will actually be configured, and which systems will be discarded without analysis.

The configuration method eliminates alternatives based on two criteria:

1. Is the combination of alterables in the alternative compatible?
2. Are any of the alterables in the alternative undesirable?

The first question is used to eliminate those alternatives that aren't logical. The second question screens out alternatives that include hardware from the database that was not

selected for a particular analysis (only three of the six flywheels in the database may be required for a particular study, etc.). Once an alternative has passed these checks, the system model can start the iterative sizing process.

2.2.5.3 Sizing. The sizing process is based on three sets of requirements. They are:

1. ACS requirements
 - (a) Momentum
 - (b) Torque
2. Power requirement
3. Reliability requirement

The system-level model sizes to meet these requirements, starting with ACS and working down to reliability. Specific implementations are discussed below.

Meeting ACS Requirements. An initial sizing attempt is made based on the number of flywheels the attitude control system requests. The attitude control system initial guess is based primarily on energy required to meet attitude control and energy storage requirements simultaneously. For non-IPACS designs, this is two flywheels. For IPACS/GFA designs, it is at least three; for IPACS/MWA, four. It has no bearing on TACS/B systems. IPACS designs are also required to meet torque and momentum requirements. The system model sizes to meet these requirements by adding wheels, one at a time, until the requirements are met. TACS may or may not meet the requirement. It is assumed that if TACS are to be compared to IPACS then the analyst would put mission-compatible TACS hardware into the database.

Meeting Power Requirements. System power requirements are based on the power load, selected hardware, and mission profile. The energy storage system is sized to meet the energy storage required to operate all systems for the maximum eclipse

duration. Equation 2.100 is used to determine the total energy storage requirement.

$$E_{required} = E_{max\ eclipse} + E_{ESS\ housekeeping} + E_{ACS\ requirement} \quad (2.100)$$

The number of units required to meet the energy storage need is given by Equation 2.101.

$$n_{ES\ Units} = \frac{E_{required}}{Unit\ Capacity} \quad (2.101)$$

This number must be an integer, so some adjustment is also required. To make the result of Equation 2.101 an integer the answer is first truncated. The resulting integer is incremented by one for an IPACS, by two for a TACS/FES system and by one for a TACS/B system. If this final result is larger than the number of wheels required to meet the ACS requirements, it becomes the new minimum number of wheels required for the system.

Meeting Reliability Requirements. Sizing for reliability is accomplished by evaluating the system reliability equation and comparing the result to the reliability requirement. If the system reliability is less than the requirement, additional flywheels are added until the reliability requirement is met or until the system reliability stops improving. Reliability sizing is not performed on systems that use TACS. TACS are ignored because the reliability sizing algorithm would increase ESS reliability to senseless levels to compensate for the fixed TACS reliability. This would tend to bias against conventional attitude control systems and produce an unrealistic trade.

2.2.5.4 Performance. After the system is sized to meet requirements, the system performance measures need to be calculated. These performance measures are:

1. **Energy Storage Performance** This is the ratio of energy storage capacity to the energy storage required. A value of 1.0 indicates that the energy storage system meets the given requirement.
2. **Attitude Control Performance** This performance measure is a weighted sum of the torque capability and the momentum storage capacity.

3. **Cost** This measure is the sum of the non-recurring and recurring costs for the alternative design.
4. **Mass** This measure is the ratio of the PACS mass to the system dry mass.
5. **Volume** This measure is the total PACS volume
6. **Reliability** This measure is the overall reliability of the PACS

Detailed methods are presented next.

Energy Storage Performance. Energy storage performance is computed using Equation 2.102.

$$P_{Energy\ Storage} = \frac{n_{ES\ Units} \times Unit\ Capacity}{E_{Required}} \quad (2.102)$$

The number of energy storage units ($n_{ES\ Units}$) is driven by ACS, ESS and reliability requirements. This causes the range for $P_{Energy\ Storage}$ to be greater than or equal to 1.0.

Attitude Control Performance. Attitude Control performance is calculated using Equation 2.103.

$$P_{Attitude\ Control} = 0.25 \times (Momentum\ Capacity) + 0.75 \times (Torque\ Capacity) \quad (2.103)$$

This composite performance measure for attitude control has no physical meaning and is used only to discriminate between alternative designs. The underlying properties, torque and momentum storage capacity, are calculated in the hardware model for attitude control.

Cost. The system cost is given by Equation 2.104.

$$P_{Cost} = \sum Subsystem\ Cost_{non-recurring} + n_{manufactured} * \sum Subsystem\ Cost_{recurring} \quad (2.104)$$

The subsystem recurring and non-recurring costs are calculated in the hardware component or subsystem models and reported in then-year dollars inflated or deflated to a specified base year (18). The current model supports fiscal years 1992 through 2005.

Mass. The system mass is calculated using Equation 2.105.

$$P_{Mass} = \sum Subsystem\ Mass \quad (2.105)$$

The subsystem masses are calculated in the hardware and subsystem models.

Volume. The system volume is calculated using Equation 2.106.

$$P_{Volume} = \sum Subsystem\ Volume \quad (2.106)$$

The subsystem volumes are calculated in the hardware and subsystem models.

Reliability. The system reliability is determined by using Equation 2.107

$$R_{system} = R_{ESS} \times R_{ACS} \times R_{PMAD} \times R_{EGS} \times R_{catastrophe} \quad (2.107)$$

The terms in equation 2.107 are dependent on the particular system configuration. R_{PMAD} is the reported end-of-life reliability for the PMAD system. R_{EGS} is the reported end-of-life reliability for the solar array. $R_{catastrophe}$ is the probability of avoiding catastrophic flywheel failure resulting in the loss of the satellite. R_{ESS} is a k out of n reliability problem, where k is the minimum number of energy storage units required to meet the energy storage need and n is the total number of energy storage units on board. In the case of a TACS this reduces to a simple series reliability calculation ($k = n$). For IPACS-type systems it is possible that there may be some degree of redundancy ($k \neq n$). Finally, R_{ACS} is either a simple end-of-life reliability (TACS) or a k out of n system (IPACS). Equation 2.108 is the general form of the k out of n .

$$R = \sum_{i=k}^n \binom{n}{i} p^n (1-p)^{n-i} \quad (2.108)$$

The catastrophic failure model is based on whether or not the system contains flywheels as components. The risk of catastrophic failure is proportional to the number of flywheels in the system.

$$R_{catastrophe} = 1.0000 \quad (2.109)$$

$$R_{catastrophe} = (0.9999)^{n_{flywheels}} \quad (2.110)$$

Equation 2.109 is used when flywheels are not in the alternative being evaluated, while Equation 2.110 applies in all other cases.

The performance measures computed with the preceding equations are used to calculate the objective measures. These are discussed next.

2.2.5.5 Utility. The objective measures are computed by transforming the performance measures using a utility equation. As implemented in this model, utility is a value on a zero to one scale which indicates how much a decision-maker values a particular level of performance. The decision-maker's value increases with increases in performance up to a certain level, beyond which he is indifferent to further improvement. This level of performance receives a utility of one; higher performance scores receive the same utility score. At the other end of the spectrum is the point of minimum utility. This is the point at which the performance is minimally useful to the decision-maker and scores a zero; lower performance will also score zero utility.

Utility profiles are required for each of the performance measures. These profiles specify the points where utility is maximized or minimized, as well as several intermediate points. The model interpolates between these points to compute the utility scores for a particular performance measure. The utility scores become the objective measures (Equation 2.111) that are used to compute the value score of the alternative.

$$\bar{O} = U(\bar{P}) \quad (2.111)$$

where \bar{O} is the vector of objective measures and \bar{P} is the vector of performance measures. U is the utility transformation function.

2.2.5.6 Value. The value score is the overall ranking of the system. It is based on the weights provided by the decision-maker and the computed objective measures. It is given by Equation 2.112.

$$V = \bar{w}^T \bar{O} \quad (2.112)$$

where \bar{w} is a vector of the decision weights and \bar{O} is the vector of objective measure values for the system being evaluated.

2.2.5.7 Optimization. The optimal design is selected by sorting the list of alternate designs according to their reported value score. The model does not currently optimize individual designs for value; it merely sizes alternative systems to meet or exceed all specified requirements. Specific design optimization (internally optimizing the optimal design to squeeze the highest possible value score from it) could be accomplished by manually changing the number of flywheels and observing the change in the value score to determine its peak value. For the purposes of our study we are limiting optimization to an optimal allocation of resources to functions (which alternative best meets the design requirements) and sensitivity analysis of the optimal result (how robust is the optimal design).

2.3 Analysis Approach

We began our analysis with four goals in mind. The first goal was to analyze how well flywheels perform in comparison to current and near-future battery technologies. To accomplish this goal, we defined a design space to evaluate flywheels with respect to batteries. Our second goal was to evaluate flywheels for a specific mission. These missions were chosen to be representative of typical space missions. Next, we wanted to evaluate how changes to mission data affect the results. To accomplish this goal, we performed a sensitivity analysis on the results with respect to changes in mission data. Since flywheels for use in the space environment are still in the development stage, our final goal was to

examine how changes to flywheel data affect our results. To accomplish this we performed a variability analysis on the results with respect to changes in key flywheel parameters.

2.3.1 Experimental Procedure. To achieve our purpose, we incorporated the ability to change both mission and flywheel data in our model. The following outlines how we achieved our four goals.

Design Space Exploration We wanted to consider how well flywheels compared to batteries in a general range of space missions. Therefore, we developed a design space in which each point represents a possible mission. Our design space was composed of key mission parameters that had the greatest effect on the decision-making process. To define the key parameters, we examined the model inputs and came up with six categories of inputs. These categories are the ones we felt most affected decision-making for space missions. The six categories are as follows:

1. Power load profile
2. Mission duration
3. Orbital altitude
4. Spacecraft size
5. Utility functions
6. Objective weights

By varying the values of the key parameters within each category, we examined how flywheels perform throughout a large range of inputs. We limited each category to no more than three different choices as a large number of inputs in each category would quickly increase the number of runs necessary to fully evaluate the design space. The design space consisted of a selection from each of the six categories. Our intent was to discover trends within the design space with respect to the key parameters.

Mission Analyses Our mission analyses compared FES to batteries. We evaluated two typical mission profiles, a GEO Meteorological Satellite (GMS) and a LEO Infrared Satellite (LIRS).

Variability Analysis We performed variability analysis on the two point analyses to examine the effects changing flywheel data had on the results. Since flywheels are still in development, our data represents only estimates. By examining the effect changes cause, we can see which flywheel parameters are most important for incremental gains.

Sensitivity Analysis We used sensitivity analysis to determine the stability of our top answer for each of the two mission analyses. We examined the effect they had on the results by changing selected mission inputs incrementally,. This allowed us to determine:

1. How each particular mission variable affects the optimal design
2. Which hardware components are impacted by which mission variables
3. How sensitive the optimal design is to changes in its environment

This factor-by-factor analysis allowed us to quantify, around a design point, which design space variables drive the selection of which pieces of hardware. We accomplished sensitivity analysis by varying

1. Power load profile
2. Spacecraft size
3. Objective weights
4. Design Life

over small ranges from the points used in the mission analyses.

These four areas combine to give us:

1. A map of the design space which can be used to make preliminary design trades or to identify which technologies should be considered for a particular mission.
2. An assessment of the validity of the model based on real-world data and mission requirements.
3. Design recommendations with quantifiable statements of confidence.

4. An assessment of the potential variability of the best answer reported by the model.
5. A analysis of hardware and alternative trends based on identifiable factors in the general design space and sensitivity to small changes in those factors around specific points.

2.3.2 Data Collection. The data was split into two basic categories: data that was varied to examine the effect on the results and data that remained constant throughout all of the model runs. The data that remained constant is presented in Tables 2.8-2.13. The data that was varied is presented in Tables 2.14-2.19 and Figures 2.5-2.11.

2.3.2.1 Constant Data. We examined three different sized flywheels. The flywheel data remained constant throughout the model runs, except for those factors changing during variability analysis. The data is presented in Table 2.8. All other flywheel data were calculated from this data. Because there are currently no standard flywheel designs or specifications, all our flywheel data is derived from the flywheel SatCon developed for Phillips Laboratory (11). Our large flywheel is directly based on SatCon's flywheel specifications. The medium and small flywheels are scaled back versions of our large flywheel. The flywheels are scaled such that their capacities are comparable to the capacities of batteries we are evaluating. We increased the mass and volume by approximately 10% for the launch-rated versions of these flywheels. We choose 10% as a representative value to account for increase in the size and mass of the magnetic bearings or addition of physical bearings and a pressure vessel around the flywheel.

The battery data also remained constant throughout the model runs. We examined two different types of batteries, Ni-H₂ and Li-Ion. Ni-Cd was rejected from consideration based on earlier model runs and a direct comparison of its characteristics with those of Ni-H₂. Li-Ion was only considered for GEO missions. Three Ni-H₂ and two Li-Ion batteries were considered. The data is presented in Table 2.9. Battery data was taken from a number of references (7), (15), (6), (5), (4), (8), (23). All other battery data was calculated from this data.

Table 2.8 Flywheel Data

| | units | Large Flywheel | Medium Flywheel | Small Flywheel |
|-------------------------------------|-------------------|----------------|-----------------|----------------|
| Moment of Inertia | kg m ² | 1.46 | 0.4 | 0.08 |
| Maximum Wheel Speed | RPM | 40,000 | 50,000 | 60,000 |
| Minimum Wheel Speed | RPM | 13,333 | 18,000 | 20,000 |
| Motor Efficiency | | 0.9 | 0.92 | 0.93 |
| Generator Efficiency | | 0.9 | 0.92 | 0.93 |
| Controller Power | W | 35 | 30 | 25 |
| Bearing Power | W | 15 | 10 | 5 |
| Mass | kg | 68 | 41 | 20 |
| Volume | m ³ | 0.115 | 0.035 | 0.011 |
| Additional Mass for Launch Rating | kg | 7 | 4 | 2 |
| Additional Volume for Launch Rating | m ³ | 0.01 | 0.004 | 0.001 |
| Reliability | | 0.95 | 0.95 | 0.95 |
| Rotor Control Error | | 0.01 | 0.01 | 0.01 |
| Motor Torque | Nm | 0.9 | 0.3 | 0.05 |

Table 2.9 Battery Data

| | units | Ni-H ₂ Large | Ni-H ₂ Medium | Ni-H ₂ Small | Li-Ion Medium | Li-Ion Small |
|----------------------|----------------|----------------------------|-----------------------------|----------------------------|------------------|-----------------|
| Rated Capacity | A hr | 90 | 40 | 12 | 50 | 20 |
| Mass | kg | 50.4 | 22.3 | 11.35 | 13.432 | 7.36 |
| Volume | m ³ | 0.04747 | 0.03288 | 0.01424 | 0.006624 | 0.003625 |
| Charge Efficiency | | 0.92 | 0.92 | 0.92 | 0.965 | 0.965 |
| Discharge Efficiency | | 0.92 | 0.92 | 0.92 | 0.965 | 0.965 |
| Reliability | | 0.999 | 0.999 | 0.999 | 0.999 | 0.999 |

The same solar array material, gallium arsenide, and configuration, panel-mounted array, was used for all model runs. The data, from (16), is presented in Table 2.10.

Table 2.10 Solar Array Data (Gallium Arsenide)

| | |
|--------------------------|------------------|
| Array Efficiency | 0.18 |
| Array Specific Power | 25 W/kg |
| Array Annual Degradation | 0.0275 |
| Array Temp Inefficiency | 0.0027 per deg C |
| Array Design Efficiency | 0.79 |

PMAD efficiencies remained constant for all model runs. The data, from (31), is presented in Table 2.11.

Table 2.11 PMAD Efficiencies

| | | | |
|----------------|------|----------------|------|
| BCC Efficiency | 0.95 | BDC Efficiency | 0.92 |
| PPT Efficiency | 0.9 | PCM Efficiency | 0.92 |
| PDU Efficiency | 0.95 | PCG Efficiency | 0.95 |

We held the TACS data constant. Six TACS were modeled, of varying sizes and methods. The first four are representative of momentum wheel configurations, and the last two are representative of control moment gyro configurations. Most of the data is from current manufacturers (14), (3). Cost data was estimated as actual numbers were not readily available. The data is presented in Table 2.12.

Table 2.12 Attitude Control Data

| | units | TACS 1 | TACS 2 | TACS 3 | TACS 4 | TACS 5 | TACS 6 |
|----------------------|----------------|--------|--------|--------|--------|--------|--------|
| Mass | kg | 8.9 | 14.3 | 6 | 10.5 | 115 | 140 |
| Non-recurring Cost | M\$ | 1.5 | 2.0 | 1.25 | 1.9 | 10.0 | 12.0 |
| Recurring Cost | k\$ | 100 | 150 | 80 | 125 | 300 | 350 |
| Volume | m ³ | 0.015 | 0.03 | 0.005 | 0.021 | 0.25 | 0.37 |
| One Year Reliability | | 0.9921 | 0.9921 | 0.9921 | 0.9921 | 0.99 | 0.99 |
| Torque | Nm | 0.08 | 0.296 | 0.04 | 0.3 | 304 | 440 |
| Momentum Storage | Nms | 27 | 81 | 16.6 | 50 | 304 | 440 |
| Operating Power | W | 13 | 28 | 10 | 40 | 15 | 15 |
| Peak Power | W | 110 | 280 | 17 | 280 | 50 | 50 |

2.3.2.2 Design Space Data. The design space exploration used a set of consistent values for certain mission variables. Launch energy is representative of the launch energy required for the Defense Meteorological Satellite Program's Block 5D-2 spacecraft (7). We generated the remaining numbers to represent a typical satellite program's requirements. The data is based on our experience and research. This mission data is presented in Table 2.13.

The data for the design space was varied by group to show the effects of changing those data points. An attempt was made to keep the amount of changing data to a

Table 2.13 Design Space - Mission Data

| | |
|-----------------------|----------|
| Launch Energy | 300 Whr |
| Design-to-Reliability | 0.9 |
| Number Manufactured | 6 units |
| Monetary Calculations | FY96 \$k |
| Slews per orbit | 2 |

minimum while keeping the inputs reasonable. The groups consisted of size, altitude, lifetime, power profile, objective weighting, and utility curves.

The first group of data is the size data, representative of two different sizes of spacecraft. See Table 2.14 for the data.

Table 2.14 Design Space - Size Variables

| | units | Large | Small |
|----------------------------|-------------------|--------|-------|
| Dry Mass | kg | 3,000 | 750 |
| I_x | kg m ² | 7,000 | 400 |
| I_y | kg m ² | 12,000 | 600 |
| I_z | kg m ² | 15,000 | 800 |
| Total Cross-Sectional Area | m ² | 65 | 20 |

Two different altitudes were examined, GEO and LEO. Only a single LEO altitude was examined. We linked ACS requirements with the altitude to avoid unreasonable requirements on the attitude control system. The data is presented in Table 2.15.

Table 2.15 Design Space - Altitude Variables

| | units | GEO | LEO |
|------------------|---------|--------|-----|
| Altitude | km | 35,786 | 700 |
| ACS Stability | degrees | 0.01 | 0.1 |
| ACS Slew | degrees | 1 | 30 |
| ACS Time to Slew | minutes | 2 | 12 |

Two lifetimes were examined in the design space. The values are presented in Table 2.16.

Table 2.16 Design Space - Design Life Variables

| | units | Long | Short |
|-------------|-------|------|-------|
| Design Life | yr | 15 | 5 |

Three power profiles were considered for the design space. The high peak profile represents a very large power requirement that occurs for only part of the orbit. The high average power profile represents a constant, large power requirement for the entire orbit. The low average power profile represents a smaller, constant power requirement for the entire orbit. An early test of the model helped guide our choices of power profiles, including using two different power profiles based on the size of the spacecraft. The early test used the same power load profiles for both large and small spacecraft. The high peak profile was also much larger. The early results showed that 2000 W was unreasonable for a small spacecraft as the mass ratios were over 1. The initial high peak profile also created unreasonable mass ratios for both large and small spacecraft. As a result, the power load profiles were changed to reflect the data presented in Table 2.17 and Table 2.18. These early results were also a factor in the choice to use mass ratios rather than a direct measure of mass.

Table 2.17 Design Space - Power Profile (Large Spacecraft)

| | High Peak | | High Avg | | Low Avg | |
|--------|-----------|----------------|----------|--------------|----------|--------------|
| | Load (W) | Period (min) | Load (W) | Period (min) | Load (W) | Period (min) |
| Load 1 | 4000 | 40 | 2000 | T_{Orb} | 500 | T_{Orb} |
| Load 2 | 1200 | $T_{Orb} - 40$ | N/A | | N/A | |

Table 2.18 Design Space - Power Profile (Small Spacecraft)

| | High Peak | | High Avg | | Low Avg | |
|--------|-----------|----------------|----------|--------------|----------|--------------|
| | Load (W) | Period (min) | Load (W) | Period (min) | Load (W) | Period (min) |
| Load 1 | 1250 | 40 | 750 | T_{Orb} | 250 | T_{Orb} |
| Load 2 | 500 | $T_{Orb} - 40$ | N/A | | N/A | |

Raw scores for the performance measures are converted into utility scores. The six performance measures are mass, cost, volume, reliability, energy storage, and attitude control. To generate the utility curves for mass, cost, and volume, we ran the model and identified the output ranges. From this data we developed utility curves that encompassed the full range of reasonable output values. Once the range was developed, points within were chosen based on a mostly linear relationship. Two types of utility curves were developed for the design space. The first is identified as the *Normal* and represents the average

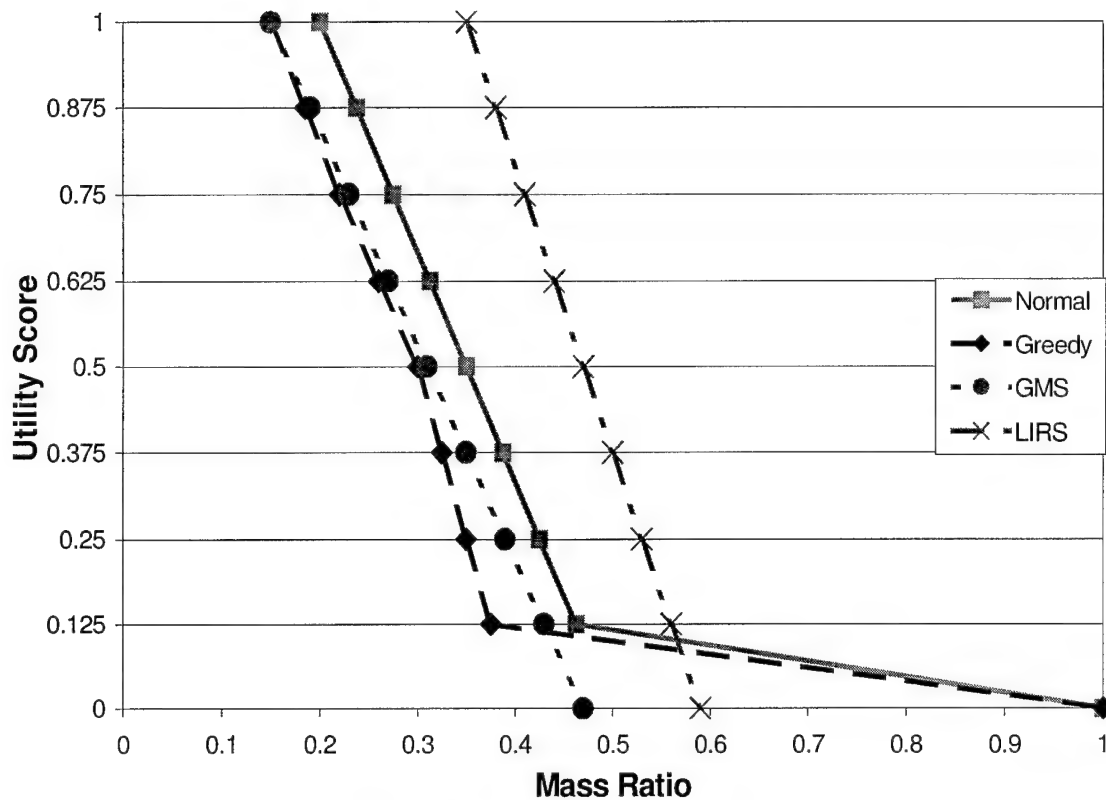


Figure 2.5 Mass Utility Curve

of our individual values. The second is the *Greedy* utility curve. The greedy curves place more emphasis on better performance.

To translate raw mass values into utility scores, the total mass of the subsystems examined is converted into a mass ratio by dividing by the spacecraft dry mass. The ratio is then converted into a utility score using Figure 2.5.

Total cost of the subsystems examined is used to develop a cost utility score. The total cost is the non-recurring cost plus the product of the number manufactured and the recurring cost. This total is converted to a utility score using Figure 2.6.

The volume of the subsystems examined is converted to a utility score using Figure 2.7.

The reliability of the subsystems examined is converted to a utility score using Figure 2.8. Points for the reliability utility transformation were chosen based on an estimation of relative values.

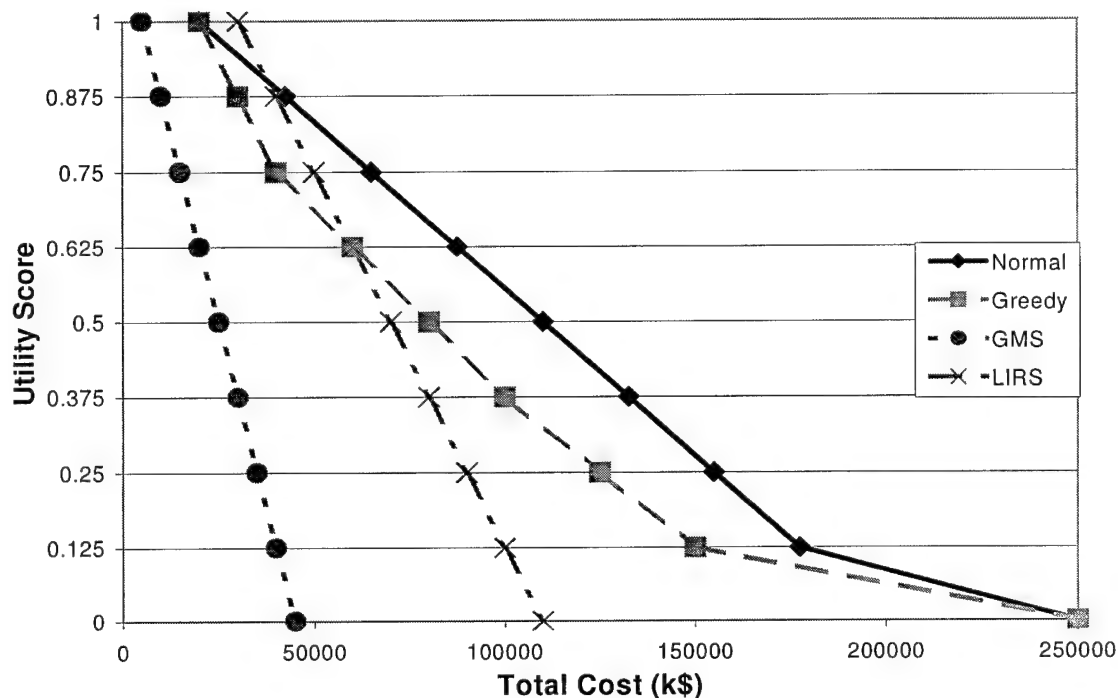


Figure 2.6 Cost Utility Curve

The ratio used to create a utility score is the ratio of energy storage available to energy storage required. Energy storage required is dependent on the alternative. The ratio is converted to a utility score using Figure 2.9. Points for the energy storage utility transformation were chosen based on an estimation of relative values. All alternatives were sized to meet energy storage requirements, hence a ratio of 1.0 was the minimum ratio possible. Its utility score was set to zero for both greedy and normal curves. The utility score is maximized when additional excess energy would provide no additional value. For the normal curve, this value was chosen to be a ratio of 1.25. Its utility score was set to 1. We designed the greedy curves to show more emphasis on performance, hence a higher performance was considered for the limiting value. This value was chosen as 1.5 and its utility score set to 1. The intervening points for both curves were chosen to properly demonstrate a typical decision-maker, which was chosen under the guidelines of

1. meet requirements and
2. diminishing returns.

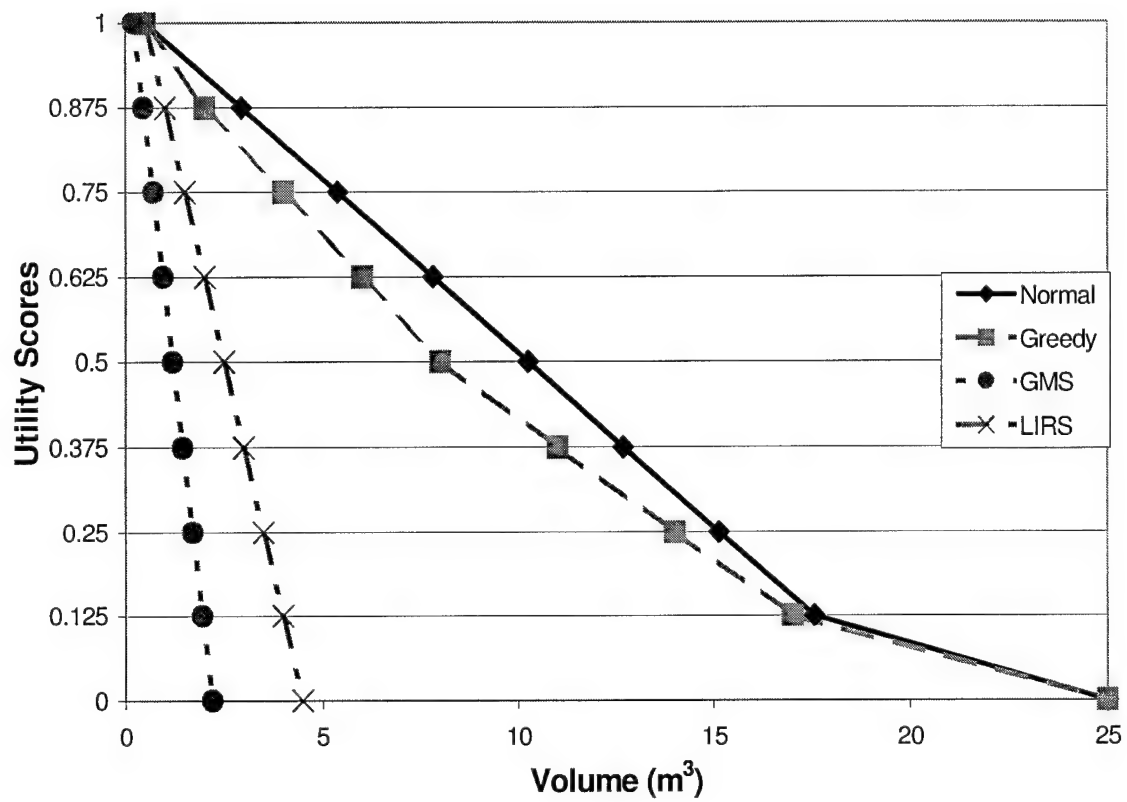


Figure 2.7 Volume Utility Curve

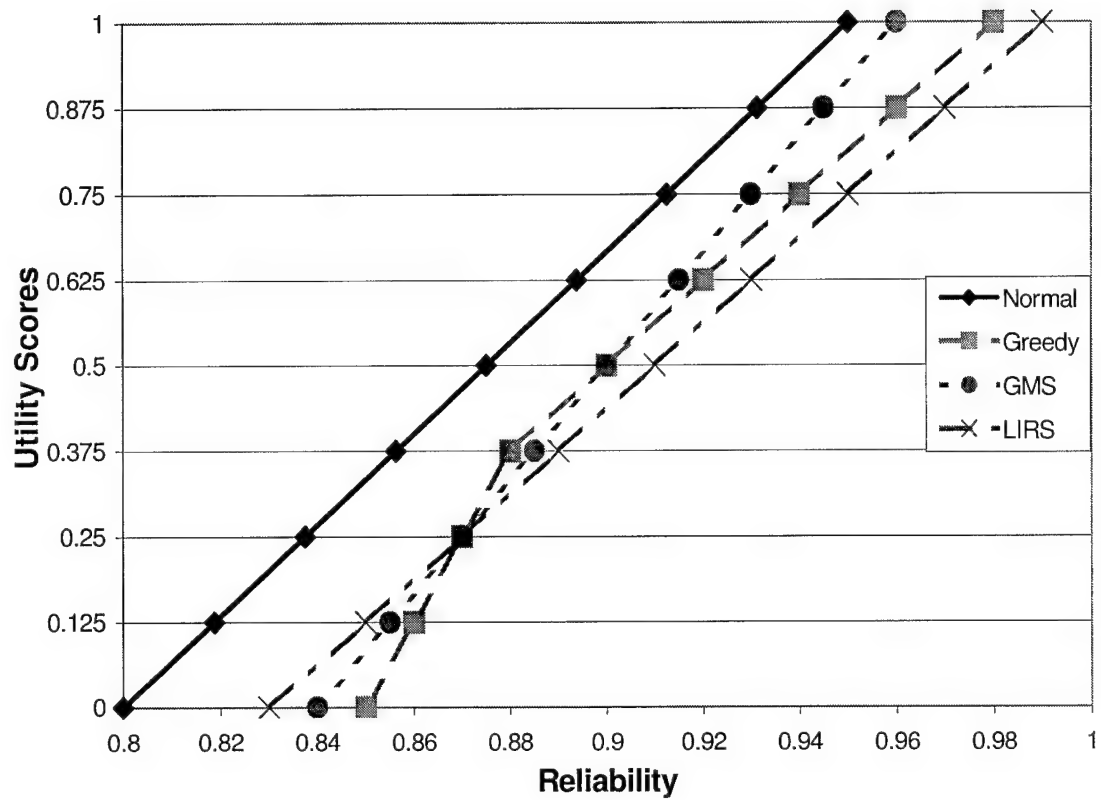


Figure 2.8 Reliability Utility Curve

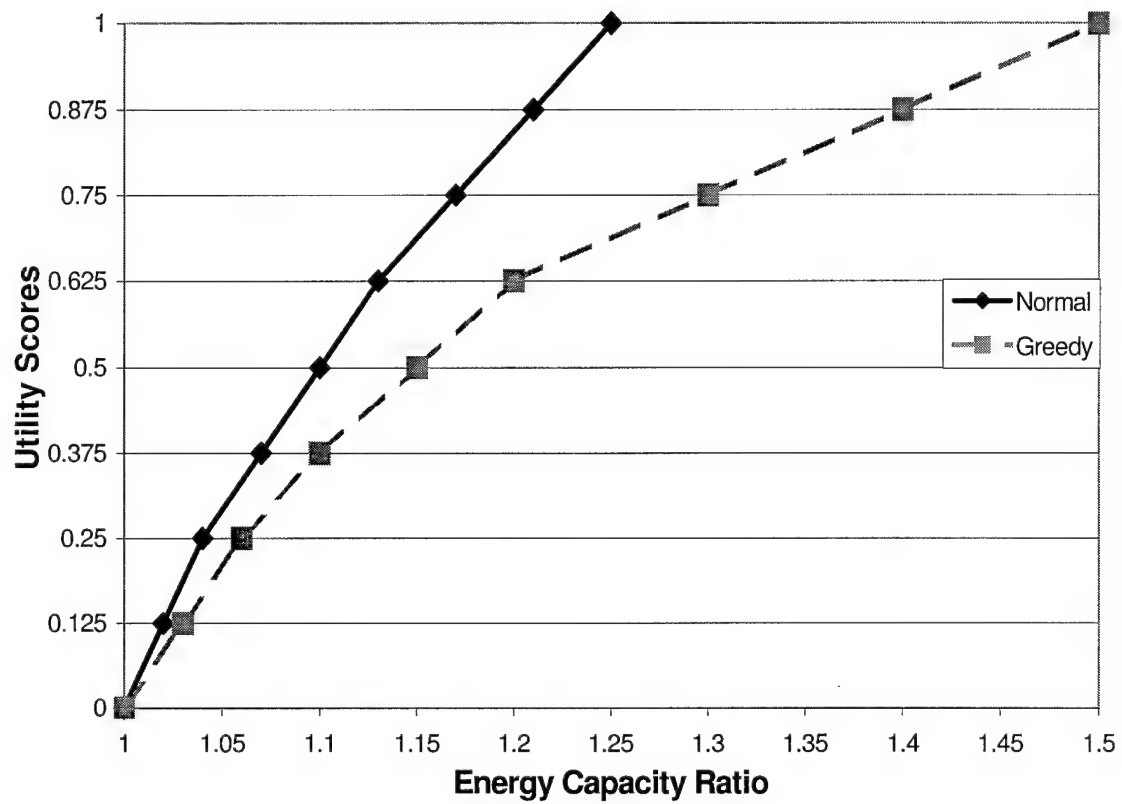


Figure 2.9 Energy Capacity Utility Curve

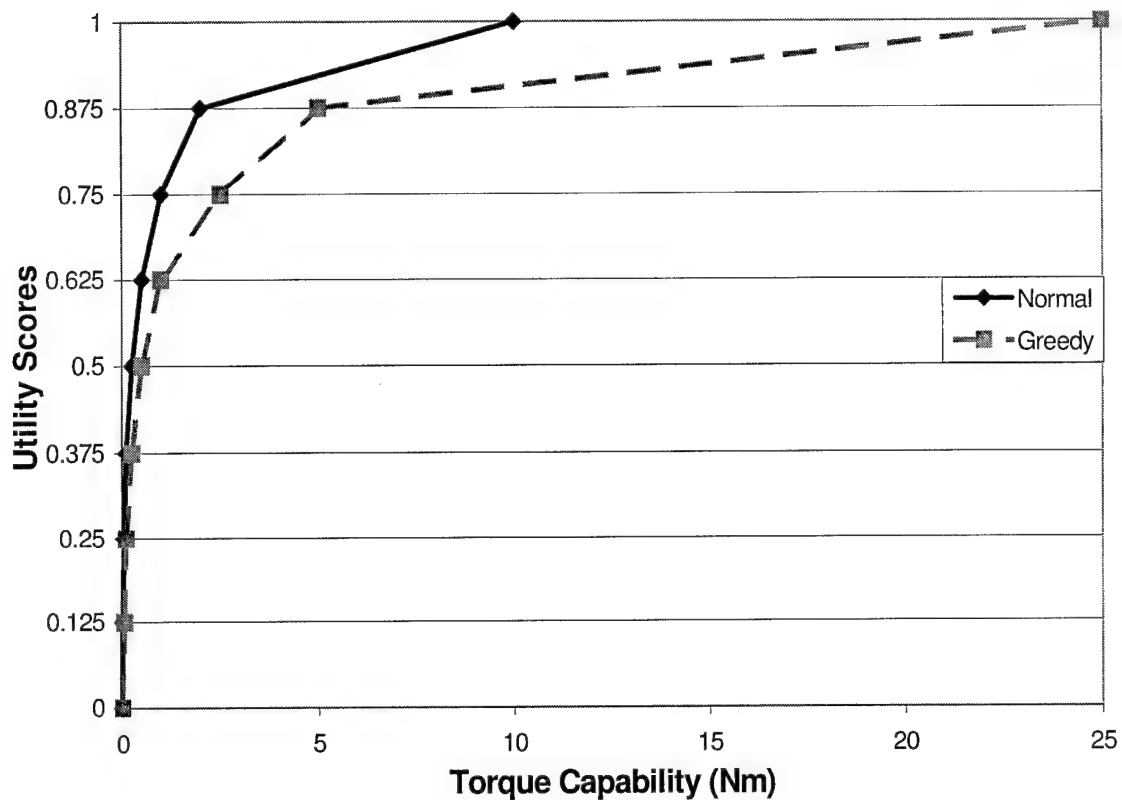


Figure 2.10 Torque Capability Utility Curve

Under these guidelines, the chosen curves resemble a mix of linear and exponential-shaped curves.

The attitude control utility score is comprised of the utility scores of the two performance measures, torque capability and momentum storage capacity in a 3:1 ratio.

The torque capability used to create a utility score is the torque capability of the attitude control system minus the required torque, which is dependent on the alternative. The result is then converted to a utility score using Figure 2.10. The curves were chosen to reflect diminishing returns, with high values possible, but most of the value achieved early. This clearly reflects an exponential curve, but the points are not from a specific exponential equation.

The momentum storage capacity used to create a utility score is the momentum storage of the attitude control system minus the required momentum storage, which is dependent on the alternative. The result is then converted to a utility score using Fig-

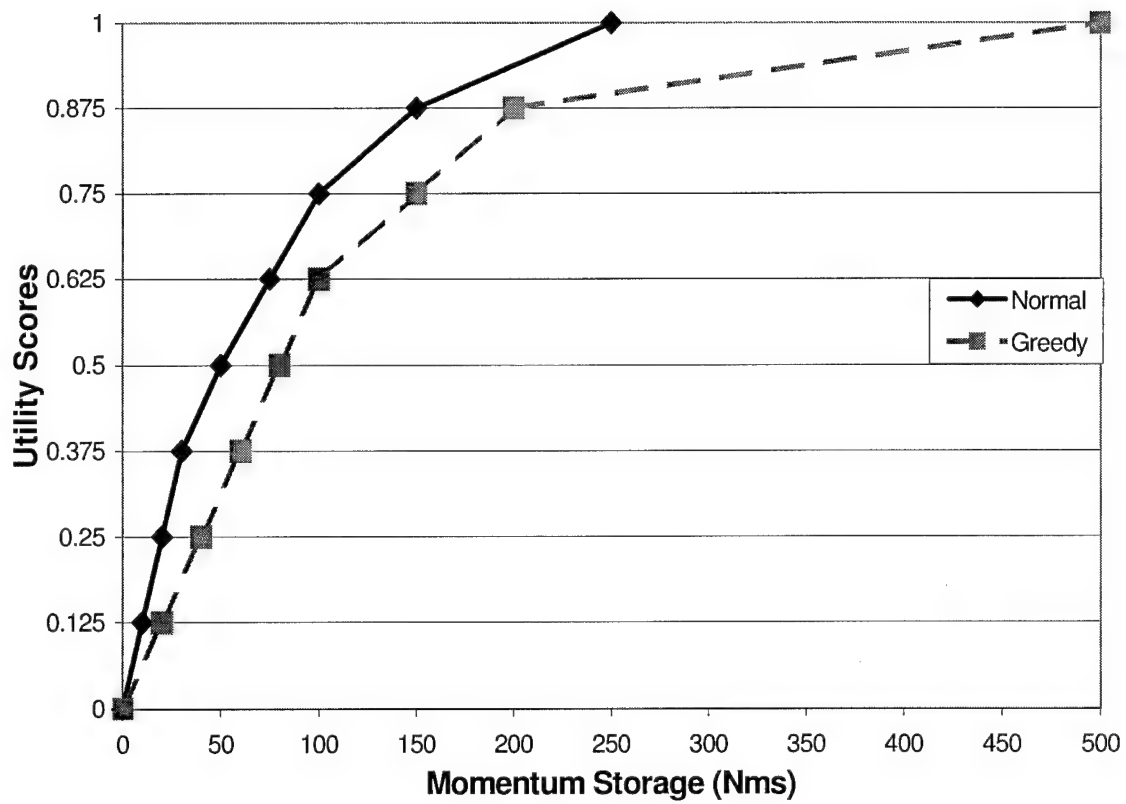


Figure 2.11 Momentum Storage Capacity Utility Curve

ure 2.11. The curves were chosen to reflect diminishing returns, with high values possible, but most of the value achieved early. This clearly reflects an exponential curve, but the points are not from a specific exponential equation.

Table 2.19 Design Space Data - Weighting Variables

| | Cost | Performance | Reliability |
|-----------------------|-------|-------------|-------------|
| Mass weighting | 0.233 | 0.1 | 0.125 |
| Cost weighting | 0.233 | 0.1 | 0.125 |
| Volume weighting | 0.233 | 0.1 | 0.125 |
| ESS weighting | 0.1 | 0.3 | 0.125 |
| ACS weighting | 0.1 | 0.3 | 0.125 |
| Reliability weighting | 0.1 | 0.1 | 0.375 |

The weights are used to combine the six utility scores into a single value score for the alternative. Three combinations of weights were used to explore the effect of the weights on alternative selection. We chose these combinations to convey preference towards cost, performance, or reliability variables. The values are presented in Table 2.19.

2.3.2.3 GEO Meteorological Satellite Data. The analysis for the GMS used the flywheel and battery data from the design space study. Mission data was altered to reflect the typical GMS mission. We tested sensitivity to a specific mission category by varying the value and leaving all other mission data at the nominal value. Tables 2.20-2.24 list the nominal mission values and the values used to test sensitivity. Utility curves are presented in Figures 2.5-2.11. For energy, torque, and momentum storage capacity, GMS used the *Normal* utility curves from the design space exploration. The nominal data guided the development of the other four utility curves. We developed the weights in Table 2.24 to reflect our value system. To examine sensitivity to flywheel data, we changed four flywheel characteristics: inertia, upper speed, efficiency, and reliability. The flywheel data is presented in Table 2.25. We tested sensitivity to a specific flywheel value by varying the value and leaving all other data at the nominal value.

2.3.2.4 LEO Infrared Satellite Data. The design for the LIRS used the flywheels and batteries from the design space study. Mission data was altered to reflect a typical LIRS mission. We tested sensitivity to a specific mission category by varying the

Table 2.20 GMS Mission Data

| | |
|-----------------------|--------------|
| Altitude | 35,786 km |
| ACS Stability | 0.25 degrees |
| Slews | 0 |
| Launch Energy | 200 Whr |
| Design-to-Reliability | 0.9 |
| Number Manufactured | 1 |
| Monetary Calculations | FY96 \$k |

Table 2.21 GMS Power Profiles

| | Nominal | | +10% | | +20% | | -10% | |
|--------|----------|--------------|----------|--------------|----------|--------------|----------|--------------|
| | Load (W) | Period (min) | Load (W) | Period (min) | Load (W) | Period (min) | Load (W) | Period (min) |
| Load 1 | 1126 | 1364 | 1239 | 1364 | 1351 | 1364 | 1013 | 1364 |
| Load 2 | 400 | 72 | 440 | 72 | 480 | 72 | 360 | 72 |

Table 2.22 GMS Size Data

| Size | units | Nominal | +10% | +25% | -10% |
|----------------------------|-------------------|---------|------|------|------|
| Dry Mass | kg | 1042 | 1146 | 1403 | 938 |
| I_x | kg m ² | 150 | 200 | 250 | 135 |
| I_y | kg m ² | 800 | 880 | 1000 | 720 |
| I_z | kg m ² | 850 | 935 | 1063 | 765 |
| Total Cross-Sectional Area | m ² | 26 | 29 | 33 | 23 |

Table 2.23 GMS Lifetime Data

| Lifetime | units | Nominal | +1 | +3 | +5 | -1 |
|----------|-------|---------|----|----|----|----|
| Life | yr | 7 | 8 | 10 | 12 | 6 |

Table 2.24 GMS Weighting Data

| Value System | Nominal | Cost | Performance | Performance+ | Reliability |
|-----------------------|---------|------|-------------|--------------|-------------|
| Mass weighting | 0.25 | 0.28 | 0.24 | 0.23 | 0.24 |
| Cost weighting | 0.2 | 0.22 | 0.19 | 0.18 | 0.19 |
| Volume weighting | 0.1 | 0.11 | 0.09 | 0.08 | 0.09 |
| ESS weighting | 0.1 | 0.08 | 0.12 | 0.14 | 0.09 |
| ACS weighting | 0.15 | 0.13 | 0.17 | 0.19 | 0.14 |
| Reliability weighting | 0.2 | 0.18 | 0.19 | 0.18 | 0.25 |

value and leaving all other mission data at the nominal value. Table 2.26 - Table 2.30 list the nominal mission values and the values used to test sensitivity. Utility curves are presented in Figure 2.5 - Figure 2.11. For energy, torque, and momentum storage capacity,

Table 2.25 GMS Flywheel Data

| | | units | Nominal | 1 st Increase | 2 nd Increase |
|-----------------|-------------|-------------------|---------|--------------------------|--------------------------|
| Large Flywheel | Inertia | kg m ² | 1.46 | 1.606 | 1.825 |
| | Upper Speed | RPM | 40,000 | 44,000 | 48,000 |
| | Efficiency | | 0.9 | 0.91 | 0.92 |
| | Reliability | | 0.95 | 0.96 | N/A |
| Medium Flywheel | Inertia | kg m ² | 0.4 | 0.44 | 0.5 |
| | Upper Speed | RPM | 50,000 | 55,000 | 60,000 |
| | Efficiency | | 0.92 | 0.93 | 0.94 |
| | Reliability | | 0.95 | 0.96 | N/A |
| Small Flywheel | Inertia | kg m ² | 0.08 | 0.088 | 0.1 |
| | Upper Speed | RPM | 60,000 | 66,000 | 72,000 |
| | Efficiency | | 0.93 | 0.94 | 0.95 |
| | Reliability | | 0.95 | 0.96 | N/A |

LIRS used the *Normal* utility curves from the design space exploration. The nominal data guided the development of the other four utility curves. We developed the weights in Table 2.30 to reflect our value system. To examine sensitivity to flywheel data, we changed four flywheel characteristics: inertia, upper speed, efficiency, and reliability. The flywheel data is presented in Table 2.32. We tested sensitivity to a specific flywheel value by varying the value and leaving all other data at the nominal value.

Table 2.26 LIRS Mission Data

| | |
|-----------------------|--------------|
| Altitude | 1200 km |
| ACS Stability | 0.25 degrees |
| Slows | 2 per orbit |
| ACS Slew | 120 degrees |
| ACS Time to Slew | 5 minutes |
| Design-to-Reliability | 0.9 |
| Number Manufactured | 1 |
| Monetary Calculations | FY96 \$k |

Table 2.27 LIRS Power Profiles

| | Nominal | | +10% | | -10% | |
|--------|----------|--------------|----------|--------------|----------|--------------|
| | Load (W) | Period (min) | Load (W) | Period (min) | Load (W) | Period (min) |
| Load 1 | 1805 | 22 | 1986 | 22 | 1624 | 22 |
| Load 2 | 1625 | 33 | 1788 | 33 | 1462 | 33 |
| Load 3 | 1375 | 54 | 1513 | 54 | 1237 | 54 |

Table 2.28 LIRS Size Data

| Size | units | Nominal | +10% | -10% |
|----------------------------|-----------------|---------|------|------|
| Dry Mass | kg | 1200 | 1320 | 1080 |
| I_x | kg m^2 | 600 | 660 | 540 |
| I_y | kg m^2 | 1200 | 1320 | 1080 |
| I_z | kg m^2 | 1400 | 1540 | 1260 |
| Total Cross-Sectional Area | m^2 | 35 | 39 | 31 |

Table 2.29 LIRS Lifetime Data

| Lifetime | units | Nominal | +1 | +2 | -1 | -3 |
|----------|-------|---------|----|----|----|----|
| Life | yr | 8 | 9 | 10 | 7 | 5 |

Table 2.30 LIRS Weighting Data

| Value System | Nominal | Cost | Performance | Reliability |
|-----------------------|---------|------|-------------|-------------|
| Mass weighting | 0.25 | 0.28 | 0.24 | 0.24 |
| Cost weighting | 0.2 | 0.22 | 0.19 | 0.19 |
| Volume weighting | 0.1 | 0.11 | 0.09 | 0.09 |
| ESS weighting | 0.1 | 0.08 | 0.12 | 0.09 |
| ACS weighting | 0.15 | 0.13 | 0.17 | 0.14 |
| Reliability weighting | 0.2 | 0.18 | 0.19 | 0.25 |

Table 2.31 LIRS Launch Power

| | units | Nominal | +10% | -10% |
|--------------|-------|---------|------|------|
| Launch Power | W | 1000 | 1100 | 900 |

Table 2.32 LIRS Flywheel Data

| | | units | Nominal | High | Low |
|-----------------|-------------|-----------------|---------|--------|--------|
| Large Flywheel | Inertia | kg m^2 | 1.46 | 1.606 | 1.314 |
| | Upper Speed | RPM | 40,000 | 44,000 | 36,000 |
| | Efficiency | | 0.9 | 0.91 | 0.89 |
| | Reliability | | 0.95 | 0.96 | 0.94 |
| Medium Flywheel | Inertia | kg m^2 | 0.4 | 0.44 | 0.36 |
| | Upper Speed | RPM | 50,000 | 55,000 | 45,000 |
| | Efficiency | | 0.92 | 0.93 | 0.91 |
| | Reliability | | 0.95 | 0.96 | 0.94 |
| Small Flywheel | Inertia | kg m^2 | 0.08 | 0.088 | 0.072 |
| | Upper Speed | RPM | 60,000 | 66,000 | 54,000 |
| | Efficiency | | 0.93 | 0.94 | 0.92 |
| | Reliability | | 0.95 | 0.96 | 0.94 |

III. Data Description and Analysis

Data description and analysis consists of four sections. The design space exploration and mission analyses seek to delineate the general characteristics of the relationship between key characteristics and the selected alternative. The final two sections, variability analysis and sensitivity analysis, examine the changes that result in the model's output data when certain characteristics are allowed to vary.

3.1 Design Space Exploration

3.1.1 Purpose. The purpose of design space exploration is to survey the SLM's best alternative at a number of different points in the design space. The results of the survey provide an overview, or map, of the design space. This map is useful for a number of reasons. First, it serves as a starting point for the analysis of FES and IPACS – We use the map to provide a general estimate of the feasibility of FES and IPACS. Second, it allows for identification of general trends within the design space. Third, it provides a quick look at the sensitivity of the results. Fourth, the results of the survey highlight areas of interest in the design space. The regions of interest are those where answers provided by design space exploration are unclear or difficult to explain. Finally, it serves as a reference for satellite designers who can use the map to identify hardware combinations that may be of interest in a particular mission.

We accomplish this evaluation using controlled combinations of specific variables in the SLM. Recall that the following model input variables were selected as the alterables for the design space exploration:

1. Power load profile
2. Mission duration
3. Orbital altitude
4. Spacecraft size
5. Utility functions

6. Objective weights

The design space exploration involved bringing together all possible combinations of the above alterables into distinct missions. This produced 144 independent missions that were analyzed using the system-level model. A more detailed explanation of the design space exploration procedure is found in the previous chapter.

3.1.2 Results. The results of the 144 design space exploration runs are listed below in Figure 3.1.

3.1.2.1 Initial Validation of Model. The first step in the data analysis was to determine if the model's output was valid and logical. Based on the discrete nature of the system-level model's inputs and the stepped response of the hardware-level models' outputs, we expected the model's top answer to incorporate hardware items that best match the mission's requirements in terms of size. For example, we expected that larger flywheels would best serve larger spacecraft, with greater power and torque requirements. Smaller flywheels, on the other hand, would best serve the missions of smaller spacecraft. For the above example, Table 3.1 shows that the medium flywheel dominates as the recommended hardware 53 times (out of 72 runs) for the large spacecraft missions. On the other hand, the small flywheel is the recommended hardware for 36 of the 72 small satellite runs. The remaining 36 top answers are split among both batteries and the other flywheels.

As another example, we expected that small flywheels would be preferred for the low average power mission, while larger flywheels would be preferred for the high peak and high average power missions. For low average power missions, the small flywheel is recommended for 28 of the 48 missions. For the high average power missions, the medium flywheel is recommended for 28 of the 48 missions. Finally, for high peak power missions, the medium sized flywheel is the top flywheel for 35 of the 48 runs.

An important item of interest is that the large flywheel performed poorly in this analysis. Specifically, the large flywheel was the top alternative in only 5 cases. The energy storage capacity of this flywheel is larger than necessary for the missions reviewed. Often,

Table 3.1 Design Space Exploration Results

| | | High Peak Power | | | | High Average Power | | | | Low Average Power | | | |
|---------|-------------|-----------------|-------|-------|---------------|--------------------|--------------------------|-------------------------|---------------|-------------------|-------|-------|---------------|
| | | LEO | | GEO | | LEO | | GEO | | LEO | | GEO | |
| | | Large | Small | Large | Small | Large | Small | Large | Small | Large | Small | Large | Small |
| Greedy | Cost | M | S | M | S | S | S | Large Ni-H ₂ | Medium Li-Ion | S | S | S | Medium Li-Ion |
| | Performance | M | M | M | M | M | M | M | M | M | M | M | S |
| | Reliability | M | M | M | M | S | M | M | M | M | S | M | S |
| Long | | | | | | | | | | | | | |
| Neutral | Cost | M | S | M | S | S | S | M | Small Li-Ion | S | S | S | S |
| | Performance | M | M | M | S | M | M | M | M | M | S | M | S |
| | Reliability | M | M | M | S | S | M | M | M | L | S | L | S |
| Greedy | Cost | M | S | M | Medium Li-Ion | M | S | Large Ni-H ₂ | Medium Li-Ion | M | S | S | Medium Li-Ion |
| | Performance | M | M | M | M | M | M | M | M | M | S | M | S |
| | Reliability | M | M | M | M | M | Medium Ni-H ₂ | M | Medium Li-Ion | M | S | M | S |
| Short | | | | | | | | | | | | | |
| Neutral | Cost | M | S | M | S | S | S | M | Small Li-Ion | S | S | S | S |
| | Performance | M | M | M | M | M | M | M | S | M | S | M | S |
| | Reliability | S | L | M | S | M | Medium Ni-H ₂ | M | Small Li-Ion | L | S | L | S |
| Key | | | | | | | | | | | | | |
| S | | Small Flywheel | | | | | | | | | | | |
| M | | Medium Flywheel | | | | | | | | | | | |
| L | | Large Flywheel | | | | | | | | | | | |
| Any | | Battery | | | | | | | | | | | |

only one or two of these large flywheels are needed for energy storage. The additional mass and volume of the extra flywheels required for attitude control penalize the system utility more than the additional energy storage adds to it

Overall, this preliminary analysis provided us with an initial validation of the model's output.

3.1.2.2 High Level Analysis. The next step taken in the data analysis was to extract any meaningful high level trends from data. In our case, three clear trends emerged. The first stood out immediately. Table 3.1 clearly shows that flywheels dominate as the top answer.

Flywheels were the top answer in 131 of the 144 mission runs.

This result emphasizes the future potential of flywheels in space applications. The second key result is that the flywheel IPACS configurations outperform the TACS/FES configurations. In all 131 runs where flywheels were the top answer, they were employed in an IPACS configuration. This result is not surprising since the IPACS combines two sub-systems into one hardware package that is less massive and less costly than two separate hardware packages. Third, in all of the cases where flywheels were the top answer, the model recommended that they remain locked at launch. We attribute this to the use of a constant launch energy requirement for each of the 144 missions. The high specific energy and energy density of the Li-Ion primary batteries result in a primary battery that is less massive and less voluminous than the flywheel pressure vessel required for a spinning flywheel at launch.

3.1.2.3 Initial Sensitivity Analysis. We also set up the design space exploration to serve as a preliminary trade-off study. The goal is to identify areas where sensitivity to certain factors, or inputs, may exist. The results were used to determine which portions of the design space require further investigation. The results indicate that there is sensitivity to the decision-maker's value system, specifically the weights applied to the objectives. As mentioned earlier, we used three different value systems for this analysis. Each one focuses on one of the following objectives: cost, performance, and reliability.

Substantial sensitivity to these weights is indicated in the results. Flywheels dominate the results when performance is emphasized. In fact, batteries are never selected if the emphasis is on performance. However, when the emphasis is on cost, batteries become the top answer in several cases. Batteries also become more competitive in certain missions if reliability is most important to the mission designer. Our interpretation of this data is that the results reflect the current state of the two technologies. Flywheels represent an under-developed technology where the cost and reliability are unknown. Batteries, on the other hand, cannot currently match the performance of experimental flywheels in terms of energy density and depth of discharge.

3.1.2.4 Areas of Interest. The high level analysis approach of the design space exploration allows us to look at the design space from a broad perspective. From this vantage point, we can identify areas in the design space that require a more in-depth analysis. In our case, we identified one area that we wish to explore more closely. The initial results indicate that there is sensitivity to certain factors such as the value system.

3.1.3 Conclusions. Overall, the initial results of design space exploration are useful, and provide a basis for the analysis effort. First, the results indicate that the model's output is logical. Second, they exhibit several broad trends about the use of flywheels in space applications. The most striking trend is that flywheels dominate the design space as the top performers. Third, the results show that the output is sensitive to certain characteristics such as the decision-maker's value system. Fourth, it identifies a key topic that we decided to investigate further — the model's sensitivity. Finally, the results serve as a quick look tool for satellite designers. Table 3.1 provides a one-page summary of the best answer for the different areas of the design space. However, this high-level analysis only provides answers to general design considerations. In order to produce a more complete analysis, it is also necessary to examine the design space at specific points representative of typical satellite missions.

3.2 Mission Analyses

3.2.1 Purpose. Mission analyses were performed to evaluate how flywheels perform for two typical missions. The first mission was a GEO Meteorological Satellite based on the National Oceanographic and Atmospheric Administration's GOES mission. The satellite is designed for a geostationary orbit, and has moderate power requirements (1301 W) with a short period of lower power consumption (430 W), and no slew requirement.

The second mission considered is a LEO Infrared Satellite (LIRS) based on the Air Force's low altitude Space-Based Infrared Satellite (SBIRS-Low) design. Phillips Laboratory, our sponsor, expressed interest in this mission. Since SBIRS-Low is still in the definition stage, we created LIRS as a representative mission. It is a LEO mission with a moderate power load (1685-1255 W), a fairly high launch energy (1000 Whr), and a requirement to slew once per orbit.

3.2.2 Results. The two missions were evaluated using various flywheel, battery, attitude control, and PMAD alternatives. Both missions were evaluated using the normal objective measure weights (Table 3.2). The top ten results are shown below in Tables 3.3 and 3.4 for GMS and LIRS respectively.

Table 3.2 Normal Objective Measure Weights

| | |
|--------------------|------|
| ACS Weight | 0.15 |
| Energy Weight | 0.1 |
| Reliability Weight | 0.2 |
| Cost Weight | 0.2 |
| Volume Weight | 0.1 |
| Mass Weight | 0.25 |

3.2.3 Conclusions. The winning configuration for GMS was the Small Li-Ion battery using a DET PMAD system. However, there were several other configurations with value scores comparable to that of the winning configuration. These included some with medium Li-Ion batteries and medium Ni-H₂ batteries. Two of the top ten configurations were flywheel IPACS configurations with a small flywheel. These results suggest that the higher scoring configurations for the GMS mission will be sensitive to the performance

Table 3.3 GMS Results

| <i>ACS Scheme</i> | <i>ESS Scheme</i> | <i>Launch Scheme</i> | <i>PMAD Scheme</i> | <i>Value Score</i> |
|-------------------|-------------------|----------------------|--------------------|--------------------|
| TACS 3 | Li-Ion Small | N/A | DET | 0.7252 |
| TACS 3 | Li-Ion Medium | N/A | DET | 0.7090 |
| TACS 3 | Ni-H2 Medium | N/A | DET | 0.6982 |
| TACS 3 | Li-Ion Small | N/A | PPT | 0.6900 |
| TACS 1 | Li-Ion Small | N/A | DET | 0.6899 |
| IPACS/GFA | Small Flywheel | Locked | DET | 0.6765 |
| TACS 3 | Li-Ion Medium | N/A | PPT | 0.6737 |
| TACS 1 | Li-Ion Medium | N/A | DET | 0.6736 |
| IPACS/GFA | Small Flywheel | Spinning | DET | 0.6706 |
| TACS 3 | Ni-H2 Medium | N/A | PPT | 0.6628 |

numbers of the flywheels and batteries. The sensitivity analysis section will explore this further.

The high battery performance for GMS is due to the fact that this is a GEO mission with fairly constant power requirements and no slew requirement. Such missions do not take advantage of all the positive aspects of flywheels. The only flywheel benefit the mission makes use of is the mass savings with IPACS. Because GMS requires a relatively small number of charge/discharge cycles, the batteries can be operated at a higher DOD. With a higher DOD, fewer batteries are required. However, it is worth noting that the difference between the highest scoring battery and the highest scoring flywheel was less than 0.05.

The LIRS results show a firm dominance of flywheel IPACS at that point in the design space, and a small flywheel IPACS/GFA with a DET PMAD system is the best

Table 3.4 LIRS Results

| <i>ACS Scheme</i> | <i>ESS Scheme</i> | <i>Launch Scheme</i> | <i>PMAD Scheme</i> | <i>Value Score</i> |
|-------------------|-------------------|----------------------|--------------------|--------------------|
| IPACS/GFA | Small Flywheel | Spinning | DET | 0.7064 |
| IPACS/GFA | Small Flywheel | Locked | DET | 0.7062 |
| IPACS/GFA | Small Flywheel | Spinning | PPT | 0.6452 |
| IPACS/GFA | Small Flywheel | Locked | PPT | 0.6450 |
| IPACS/MWA | Medium Flywheel | Locked | DET | 0.5835 |
| IPACS/GFA | Medium Flywheel | Locked | DET | 0.5834 |
| IPACS/GFA | Medium Flywheel | Spinning | DET | 0.5779 |
| IPACS/MWA | Medium Flywheel | Spinning | DET | 0.5742 |
| IPACS/MWA | Large Flywheel | Locked | DET | 0.5533 |
| IPACS/MWA | Large Flywheel | Spinning | DET | 0.5515 |

configuration. Since LIRS is a LEO mission with many charge/discharge cycles as well as a slew requirement, such a configuration was expected. Further, the results show that wheels both spinning and locked at launch have nearly the same value score. Since launch power requirements for LIRS are fairly high, larger primary batteries are required to supply it. The performance and budgetary impacts of larger primary batteries are practically equal to that of the launch-rated flywheels.

Our expectation was that flywheels would perform better in LEO missions than in GEO missions. The results of mission analyses support this expectation.

3.3 Variability Analysis

3.3.1 Purpose. Variability analysis attempts to answer the question, "How does varying certain hardware characteristics affect the respective SLM outcomes?" Since the IPACS outperformed TACS/FES configurations, we limited the analysis to a comparison of IPACS to TACS/B. The analysis approach varied flywheel reliability, efficiency, upper wheel speed, and moment of inertia in order to determine the relative effect on the value scores. This analysis was applied to both the LIRS and GMS missions. Since flywheels are a relatively new technology, we note the existence of some uncertainty in characterizing the flywheel specifications. Those specifications used in the analysis are essentially estimates based on the literature and the opinions of experts in the field.

3.3.2 GMS Variability Analysis. Since the batteries are the clear winners for this mission, we decided that the best course of action was to impose a consistent increase in the values of the flywheel variables to improve the IPACS performance. The changes made to the hardware characteristics of each of the flywheels were large enough to show variation, but not so large as to be unrealistic. They are the following:

- Reliability was increased by 0.01 and 0.02.
- Efficiency was increased by 0.01 and 0.02.
- Upper Wheel Speed was increased by 10% and 20%.
- Moment of Inertia was increased by 10% and 25%.

3.3.3 GMS Variability Analysis Results. The normal case exhibited a 0.049 difference between the highest scoring battery and the highest scoring flywheel.

The following summarizes the effect the changes had on this difference:

- Increase reliability by 0.01 and difference decreases to 0.027
- Increase reliability by 0.02 and difference further decreases to 0.017
- Increase efficiency by 0.01 and difference decreases to 0.042
- Increase efficiency by 0.02 and difference further decreases to 0.035
- Increase upper wheel speed by 10% and difference decreases to 0.047
- *Increase upper wheel speed by 20% and difference increases to 0.068*
- Increase moment of inertia by 10% and difference decreases to 0.046
- Increase moment of inertia by 25% and difference further decreases to 0.043

It is clear that none of the cases result in IPACS winning. The results do suggest that increases in reliability, efficiency, and moment of inertia show a trend toward a flywheel preference, the greatest benefit is realized when increasing reliability. However, the flywheel score suddenly drops considerably for a 20% increase in upper wheel speed. An examination of the data finds that IPACS/GFA with a small flywheel locked at launch and a DET PMAD system is the highest scoring flywheel configuration for all runs except for the 20% increase in upper wheel speed. We compared the change in utility scores of this configuration for both the 10% and 20% increases in upper wheel speed. The following are the results of this comparison:

- Mass utility increases by 0.08
- Cost utility increases by 0.02
- Volume utility increases by 0.02
- *Reliability utility decreases by 0.36*
- ACS utility decreases by 0.04

This case makes the distinction between sizing to meet requirements and sizing to optimize value score clear. We ensured the reliability of every alternative PACS design meets or exceeds 0.9. Using four flywheels for the case of an IPACS/GFA and a 20% increase in upper wheel speed achieves this reliability requirement.. However, to maximize the total value score, the optimal course of action would be to add a fifth flywheel. The result would be a nominal decrease in the mass, cost, and volume utilities and a significant increase in the reliability utility as well as excess energy storage capacity driven by the addition of the fifth wheel. This design would outscore the four-wheel alternative, but the model will not consider it.

3.3.4 LIRS Variability Analysis. Flywheels are the clear winners for the LIRS mission. Therefore, this compelled us to increase and decrease the values of the flywheel variables from nominal. The changes made to the hardware characteristics of each of the flywheels were large enough to show variation, but not so large as to be unrealistic. They are the following:

- Reliability was both increased and decreased by 0.01.
- Efficiency was both increased and decreased by 0.01.
- Upper Wheel Speed was both increased and decreased by 10%.
- Moment of Inertia was both increased and decreased by 10%.

3.3.5 LIRS Variability Analysis Results. The nominal case exhibited a 0.176 difference between the highest scoring battery and the highest scoring flywheel.

The following summarizes the effects the changes had on this difference:

- *Increase reliability by 0.01 and difference decreases to 0.136*
- *Decrease reliability by 0.01 and difference decreases to 0.164*
- *Increase efficiency by 0.01 and difference increases to 0.189*
- *Decrease efficiency by 0.01 and difference decreases to 0.162*
- *Increase upper wheel speed by 10% and difference increases to 0.177*

- Decrease upper wheel speed by 10% and difference decreases to 0.151
- Increase moment of inertia by 10% and difference increases to 0.179
- Decrease moment of inertia by 10% and difference decreases to 0.148

These results show a clear trend in favor of flywheels with increases in efficiency, upper wheel speed, and moment of inertia. An examination of the data finds that a small flywheel IPACS/GFA, spinning at launch with a DET PMAD system is the highest scoring flywheel configuration. A comparison was made of the configuration's utility scores for the nominal and 0.01 increase in reliability. The following are the results of this comparison:

- Mass utility increases by 0.107
- Cost utility increases by 0.02
- Volume utility increases by 0.009
- *Reliability utility decreases by 0.36*
- ACS utility decreases by 0.05

This configuration, with a 0.01 increase in reliability, met the requirement of 0.9 with four wheels. Thus, we have the same result as was seen with the upper wheel speed for the GMS mission.

3.3.6 Conclusions. The results of the variability analysis show that flywheel IPACS is the clear winner for the LIRS mission, and most certainly competitive in the GMS mission. For the GMS mission, the value scores for flywheels never exceeded that of batteries. For the LIRS mission, to the extent to which the flywheel characteristics were decreased, the battery scores never outperformed those of flywheels. For both missions, the results suggest the most advantageous approach to flywheel design optimization should focus on maximizing reliability and efficiency.

3.4 Sensitivity Analysis

3.4.1 Purpose. In the mission analyses section, we determined the best performing Power and Attitude Control System (PACS) designs for the GMS and LIRS

missions. The purpose of sensitivity analysis is to determine how sensitive the best performing designs are to changes in the mission profile variables associated with each mission (e.g. mission design life). The question we want to answer is, "How much of a change in certain mission profile variables, if any, will change the optimal power and attitude control system design?"

3.4.2 GMS Sensitivity Analysis. The results of the mission analyses reported the best performing PACS design for the GMS mission is the TACS/Small Li-ion battery with a DET PMAD system, which scored 0.73. The best IPACS design scored 0.68, a difference of 0.05.

As detailed in the Chapter 2, several mission variables were varied to perform the sensitivity analysis. For GMS, we modified the following mission variables while keeping all others constant:

- GMS power profile levels: +10%, +20%, -10%
- GMS size (dry mass, inertia, and cross-sectional area): +10%, +25%, -10%
- GMS design life (7 yrs): 6 yrs, 8 yrs, 10 yrs, 12 yrs

See Table 2.21 for power requirements used, and Table 2.22 for size data used.

The objective measure weightings used to evaluate each alternative design were also varied. The analyses placed additional weight on the following categories of measures:

- GMS cost (mass, cost, and volume measures)
- GMS reliability (reliability measure)
- GMS performance (ESS and ACS quality measures)
- Double the emphasis on GMS performance

See Table 2.24 for weightings used.

3.4.3 GMS Sensitivity Analysis Results. The results of the GMS sensitivity analysis on the mission variables are separated based on whether or not the design is sensitive to the variable.

3.4.3.1 Variables to Which GMS Design is Not Sensitive. For

variations in several mission variables, the TACS/B design remained the best performer. Nor did the second place IPACS design ever get close enough to the highest scoring design to merit further analysis. Below is a list of changes to mission variables to which the best performing design is not considered sensitive. The list also reports the difference between the winning TACS/B score and lower-rated IPACS score associated with each analysis. Recall the difference with no variations is 0.05.

- Decreasing GMS power – a 10% reduction in the power profile increased the TACS/B lead to 0.06. This is because batteries score even better at lower power levels. The minimum number of flywheels at lower power levels is dictated by reliability or ACS needs instead of ESS needs. Low power levels require fewer batteries, and the resulting utility savings from a reduced ESS mass and volume outweigh the ACS performance gains of an IPACS.
- GMS size – decreasing the dry mass by 10% increases the mass ratios on both designs. (As the dry mass drops, the ratio of PACS mass to the dry mass increases.) This boost in the mass ratios decreases both design scores. The TACS/B lead increased slightly to 0.06 because its mass score decreases more slowly than the IPACS score. Conversely, when size is increased by 10%, both scores increase. A 0.03 score difference results since the TACS/B score increases more slowly than the IPACS score.
- Decreasing GMS design life – lowering design life to six years also increased the TACS/B lead to 0.06. Batteries perform better with shorter cycle lives.
- GMS cost emphasis – when more emphasis is placed on cost in the scoring system, TACS/B improves its lead to 0.08. The TACS/B design is modeled to cost \$12M, much less than the IPACS's \$19M.
- GMS reliability emphasis – additional weight on reliability decreases the TACS/B lead to 0.04. It has a slightly lower reliability (0.93) than the IPACS design (0.94), and the additional emphasis on reliability allows IPACS to close the gap, if only slightly.

3.4.3.2 Variables to Which GMS Design is Sensitive. Changes to mission variables to which the GMS is sensitive include:

- Increasing GMS power – when the GMS power profile is raised to 120% of nominal, the TACS/B and IPACS designs have identical scores. Higher power levels favor IPACS – when batteries are used, both the number of batteries used and the solar array size must be increased to meet peak power requirements, boosting system mass, cost, and volume.
- Increasing GMS design life – if GMS design life exceeds 11 years, the IPACS design outcores the TACS/B. Flywheel energy storage outperforms batteries at longer design lives because its depth-of-discharge is not affected by the design life.
- Increasing performance emphasis – when the emphasis on performance was increased a second time, the IPACS score exceeded the TACS/B's. The IPACS design has much larger torque and momentum capabilities and consequently earns better performance scores.

In summary, the GMS TACS/B design outperforms the GMS IPACS by 0.05. If power requirements increase by 20%, or design life increases from 7 to 12 years, or performance is emphasized more, the IPACS design outcores the TACS/B design.

3.4.4 LIRS Sensitivity Analysis. In the mission analyses section, the best performing PACS design for the LIRS mission was a small flywheel IPACS/GFA, spinning on launch with a DET PMAD system. It scored 0.71. The best TACS/B design missed the top spot by 0.18, scoring only 0.53.

The analysis approach section in Chapter 2 details how mission variables were varied for LIRS sensitivity analysis. The following mission variables were modified one at a time while keeping all others constant:

- LIRS power profile levels: +10%, -10%
- LIRS size (dry mass, inertia, and cross-sectional area): +10%, -10%
- LIRS design life (nominally 7 yrs): 5 yrs, 9 yrs, 10 yrs

See Table 2.27 for power requirements used, and Table 2.28 for size data used.

We also varied the objective measure weightings used to evaluate each alternative design. Four analyses placed additional weight on the following categories of measures:

- LIRS cost (mass, cost, and volume measures)
- LIRS performance (ESS and ACS quality measures)
- LIRS reliability (reliability measure)

See Table 2.30 for weightings used.

3.4.5 LIRS Sensitivity Analysis Results. The LIRS power and attitude control design is not sensitive to any of the variables used for sensitivity analysis. No changes to the mission data even came close to cutting the IPACS 0.18 lead in half. Consequently, the TACS/B score was never comparable to IPACS to a degree warranting further analysis.

The list below shows how some variations in the data were able to close the gap between IPACS and TACS/B, but none very closely. Following each sensitivity analysis is the difference between the IPACS and TACS/B score in parentheses.

- LIRS power profile levels +10% (0.13)
- LIRS power profile levels -10% (0.18)
- LIRS size (dry mass, inertia, and cross-sectional area) +10% (0.21)
- LIRS size (dry mass, inertia, and cross-sectional area) -10% (0.12)
- LIRS design life (nominally seven years) of 5 years (0.12)
- LIRS design life of 9 years (0.18)
- LIRS design life of 10 years (0.18)
- LIRS cost emphasis (mass, cost, and volume measures) (0.12)
- LIRS performance emphasis (ESS and ACS quality measures) (0.13)
- LIRS reliability emphasis (reliability measure) (0.10)

In summary, the LIRS IPACS design outperforms the LIRS TACS/B by 0.18. No feasible variation in the LIRS mission data will allow the TACS/B design to outperform IPACS.

3.5 Summary

In design space exploration it was shown that the analysis effort produced reasonable results. The results indicate that the output of the SLM demonstrates several general trends concerning the utility of flywheels in space missions, the most substantial of which is the appearance of flywheels in the design space as the top performers. Also, it is shown that there are certain characteristics, such as the decision-maker's value system, to which the output is sensitive. The output's sensitivity was identified as warranting further investigation.

In the mission analyses section the goal was to evaluate how flywheels perform for two typical missions. The results suggest that the higher scoring TACS/B configurations for the GMS mission are sensitive to the performance numbers of both PACS types. Batteries exhibited better performance for GMS. This was expected since it is a GEO mission with power requirements that were roughly constant, and there were no slew requirements. Since GMS requires relatively few charge/discharge cycles, the battery DOD is higher and fewer batteries are required. However, it was shown that the difference between the highest scoring battery and flywheel was less than 0.05.

The results for LIRS exhibited a clear advantage in the use of flywheel IPACS. A small flywheel IPACS/GFA with a DET PMAD system was found to be the best configuration. As with GMS, this result comes as no surprise since the mission characteristics of many charge/discharge cycles and a slew requirement would suggest such a configuration. Also, spinning and locked wheels of a particular type had little difference in their value scores. LIRS launch power requirements are reasonably high, so in order to provide it larger primary batteries are needed. These large primary batteries come with a penalty, but these are comparable to the penalties on mass and cost for launch-rated flywheels. Thus, LEO missions would appear to gain the most from flywheel IPACS. Although the highest

performer for GEO missions was a TACS/B, the difference was not extremely significant. Therefore, these preliminary results suggest that flywheel applications in this regime should not be discouraged.

The focus in variability analysis was on varying certain flywheel hardware characteristics to determine how they affect the respective SLM outcomes. Batteries outperformed flywheels in the GMS mission. For LIRS, to the degree to which the characteristics were decreased, the battery scores never overtook the flywheel scores. The conclusion is that the best course of action for flywheel development is to maximize the reliability and efficiency.

In the sensitivity analysis section, it was concluded that if power requirements increase by 20%, or design life increases from 7 to 12 years, or performance is emphasized more, the IPACS design outscores the TACS/B. For the LIRS mission, IPACS design outperforms the LIRS TACS/B by 0.18. No feasible variation in the LIRS mission data will allow the TACS/B design to outperform IPACS.

IV. Conclusions

4.1 Summary

Our project began with the goal of answering the questions we posed in Chapter I. To help us answer these questions, we applied Hall's 7-Step Systems Engineering Process to the problem. However, during the course of our work we determined that Hall's approach was insufficient and as a result developed our own model-based SEP. Using the model-based SEP as a guide for our study, we developed the Flywheel Analysis Tool as an aid to our flywheel investigation. Overall, our work achieved its goal and we answered the questions about the use flywheels in space applications. The results indicate that flywheels could be beneficial for many space missions. However, considerable work still remains before flywheels can be incorporated as common components in satellites.

The initial questions we asked were:

- Can flywheels replace batteries for energy storage (FES)?
- Can flywheels replace batteries and traditional attitude control systems in a combined energy storage and attitude control system (IPACS)?
- When should FES systems or IPACS be used?
- When using IPACS, should the configuration include the momentum wheel approach or the gimbaled flywheel approach?
- Should flywheels be spinning at launch or should they be locked?
- What flywheel characteristics have the most effect on the performance of flywheel-based systems?
- What are areas of significance for future research?

Our analysis answers these questions.

Design space exploration answered the first five questions above for many design points. We conclude that flywheels are the recommended alternative for many regions within the design space. However, there are areas where batteries outperform IPACS.

In those cases where flywheels are the top answer, they should be employed in the dual IPACS role. The performance benefits of flywheels are best captured in this configuration. Additionally, the results recommend an IPACS/GFA configuration for most cases. IPACS/GFA configurations can meet the mission requirements with fewer units than IPACS/MWA configurations.

The amount of launch energy required influences whether or not the flywheels should be locked at launch. For low launch energy requirements, the high performance Li-Ion primary batteries can provide the launch energy with less mass and volume than launch-rated spinning flywheels. Spinning flywheels can provide the energy needed for larger launch energy requirements with less mass and volume than a larger primary Li-Ion battery.

We also answered the question about which flywheels characteristics are the most influential. In variability analysis, we showed that flywheel reliability and efficiency are the two key factors.

4.2 Recommendations for Future Research

It is important to recognize the need for further research in flywheel technology. The results of this study suggest benefits if flywheels are used in satellite design efforts. However, uncertainties exist regarding certain aspects of flywheel design and performance.

There was a concern for the apparent lack of flywheel reliability and failure mode data. Research programs have concentrated on the "rotor burst" mode of failure. Having knowledge of the wheel speed at which a rotor burst is likely to occur is sufficient to impose the fault avoidance tactic of simply staying below that upper wheel speed. Many more modes of failure exist and should be investigated. It must be noted that the fault avoidance approach is limited due to the cyclic fatigue that inevitably develops in any mechanical system that undergoes cycling. Cyclic fatigue has seen little research in comparison to rotor bursts.

Another aspect of flywheel performance that was found to be important to the optimization of designs is the flywheel's overall efficiency. In terms of control electronics and power conversion equipment there appears to be merit in using pulse width modulated

design. Also, optimization of the entire power system may be worth investigation. In particular, a spacecraft bus at the high voltage the flywheel reaches at its upper wheel speed would eliminate the need for power conversion at the flywheel input. This combined with pulse width modulated switching technology for the electronic commutation circuitry of the flywheel motor-generator may substantially increase efficiency.

This study dealt lightly with the possibility of using the flywheels to provide power during launch through deployment of the solar arrays. Research must be accomplished in order to provide a design robust enough to survive the launch environment if this concept is to be fully realized as a viable alternative to using primary batteries on ascent.

Finally, the literature has suggested that flywheels impose less thermal requirements to function properly. More research data should be gathered to properly defend this assertion.

Appendix A. First Iteration

A.1 Systems Engineering Design Methodology

A.1.1 A Systematic Approach to Problem-Solving. Defining systems engineering is difficult. For our purposes we consider *systems engineering* to be *the architecting and implementation of large, complex, interdisciplinary collections of subsystems into an organized unit which can accomplish tasks not achievable by the individual components.* (29)

The primary tool of systems engineering is a systematic process for problem solving. The exact tool used varies with the nature of the system, and is further modified by the intuition and experience of the systems engineer or engineering team assigned to the project. Within the context of this problem-solving framework, the systems engineer is faced with the task of selecting appropriate tools for:

- defining the problem
- generating requirements
- creating and evaluating alternatives
- modeling and optimizing potential solutions
- making decisions

Engineering is ultimately concerned with the optimal allocation of resources. A key focus is determining the most valuable use of these resources. Value is a subjective term, defined by a customer. This adds another dimension to the systems engineering process: *Value System Design*. Value system design takes the objectives of the design process and evaluates how well each alternative design meets the objectives. This systems engineering process (SEP) will be refined and improved as the design itself progresses.

A.1.2 Heuristics. Heuristics are verbal abstracts of experience. They serve as guidelines in decision making, value judgments and assessments. We have included heuristics in our design methodology to remind us of the philosophical and nonquantifiable

aspects of our design process. Some of these heuristics are goals or objectives for our design that we could not include as firm needs or objectives. Others are heuristics from (27) or from personal experience. We believe these can be applied to our design study. They are as follows:

- Minimize the complexity of the system. Keep the system as simple as possible.
- Minimize the number of subsystems needed for energy storage and attitude control.
- Relationships among the elements are what give systems their added value.
- Success is defined by the beholder, not by the architect.
- No complex system can be optimum to all parties concerned, nor all functions optimized.
- A model is not reality.
- If you can't explain it in five minutes, either you don't understand it or it doesn't work.
- The eye is a fine architect. Believe it.
- Group strongly related elements together and separate unrelated elements.
- Many requirements can be combined in ways that complement each other in the total design solution.

A.1.3 The Spiral Model. The spiral model is one of the paradigms used to describe system development. The model captures development of a system as an incremental process. Design implementation proceeds from a crude set of specifications to a successively more refined design. A stable design emerges at periodic intervals. The design is presented to the customer. The customer provides feedback to be incorporated into the next refinement cycle of the design. The spiral model minimizes the amount of rework the team will perform in coming to a final design.

A.1.4 Hall's Approach. Hall's Approach is one of the primary design methodologies in use for systems engineering. Developed by Arthur D. Hall, III (9), this approach to the SEP is an iterative application of seven steps. The steps are:

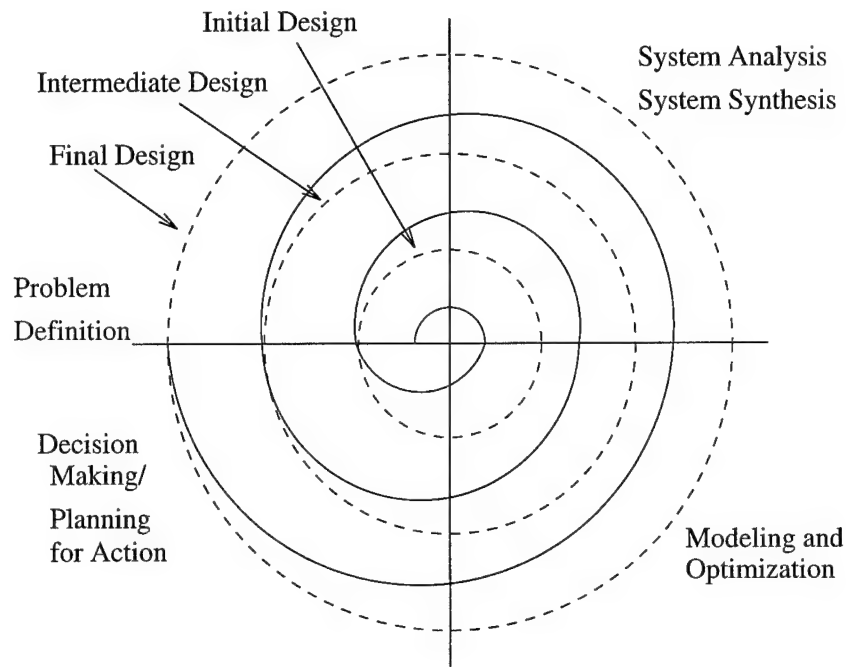


Figure A.1 The Spiral Model of System Design

1. Problem Definition
2. Value System Design
3. System Synthesis
4. System Analysis
5. Optimization (including modeling)
6. Decision Making
7. Planning for Action

This approach is general in nature and must be adapted to the specific problem. Lessons learned during successive iterations drive this adaptation.

The following sections of this Appendix detail each of Hall's seven steps and how they were applied to the design study.

A.2 Problem Definition

A.2.1 Purpose. The purpose of the Problem Definition step is to clearly identify the problem being solved; the system being designed; the people and organizations involved; and the needs, alterables, and constraints affecting the overall design.

A.2.2 The Products of Problem Definition. Sage (29) describes several products of Problem Definition. Nine of them are listed below.

1. A well conceived title for the problem
2. A descriptive scenario explaining the nature of the problem and how it came to be a problem, presenting as much history as possible with the available resources
3. An understanding of the disciplines and professions relevant to the problem
4. An assessment of scope
5. A determination of the societal sectors involved
6. An identification of the actors involved in the problem solving situation
7. An identification of *needs*
8. An identification of *alterables*
9. An identification of major *constraints*

We produced these products of Problem Definition for our system study. They are identified and described below.

A.2.3 Applying the Approach.

A.2.3.1 Title. Satellite Integrated Power and Attitude Control System Design Study

A.2.3.2 Scenario. Current Earth-orbiting satellites use solar arrays for primary power and chemical batteries to store energy for use during eclipse periods. Satellite load demands for electrical power are mission dependent. Batteries are required to supply electrical power during periods of spacecraft eclipse for a certain number of

charge/discharge cycles over the satellite's lifetime. The frequency and duration of the eclipses a mission encounters varies with orbital altitude and inclination. Typical eclipse seasons for geostationary orbits last for 45 days and occur twice a year. The maximum eclipse duration for geostationary satellite orbits is 72 minutes. Eclipse seasons for LEO satellites are much more dependent on altitude and inclination. They can vary from two weeks per year up to the entire year for some satellites. For a typical 100 minute LEO period, the maximum eclipse duration is 36 minutes.

Depth-of-discharge refers to the percentage of total battery capacity that is removed during a discharge period. For traditional aerospace chemical batteries, battery cycle life decreases with increasing depth-of-discharge. A longer satellite lifetime requirement dictates that a lower depth-of-discharge be used for the batteries. Therefore, to meet energy storage requirements over the satellite lifetime, batteries are oversized. Thus, there is an imposition of additional mass with no proportionate benefit.

A Flywheel Energy Storage (FES) system does not suffer from decreasing cycle life with higher depths-of-discharge. An FES system stores energy mechanically in the form of a rotating flywheel instead of chemically. An FES system does not need to be oversized. The excess mass and inaccessible capacity associated with chemical batteries can be avoided. These flywheels can be designed with a higher specific energy than batteries. An FES system may have less mass than chemical batteries while satisfying the same energy storage requirement.

Devices incorporating rotating wheels are used extensively onboard satellites in attitude control systems to absorb or condition satellite angular momentum. Attitude control is accomplished either by differential rotation of a flywheel array, or by applying external torques to the flywheels through an actuator and gimbal assembly. Rotational speeds on the order of 2,000 to 4,000 RPM are common. When the wheels become "saturated," they have reached the specified limit of their rotational speed. At this point, it becomes necessary to apply external torques in order to transfer the excess momentum out of the system. This is accomplished by expending propellant through the reaction control system, or by using a magnetic or gravitational torquer (where feasible). Many satellites reach their

end-of-life when they have exhausted their propellant supply and can no longer maintain the required attitude.

The flywheels used in an FES system can replace the reaction and momentum wheels used for attitude control. The resulting system is called an Integrated Power and Attitude Control System (IPACS). By achieving higher angular speeds than traditional reaction and momentum wheels, flywheels can be made with a larger angular momentum storage capacity. This may lead to less frequent use of the reaction control system to diminish accumulated momentum. Therefore, a reduction in propellant use could extend the life of the satellite.

The problem of determining the characteristics of a combined electrical power system and ACS architecture that optimizes the mission requirements is addressed in this study.

The analysis is performed on the following architectures:

- separate power and ACS using batteries for energy storage and traditional ACS
- separate power and ACS using flywheels for energy storage and traditional ACS
- flywheel IPACS

A.2.3.3 Relevant Disciplines/Professions. Some of the disciplines involved are:

- Attitude Control – design and analysis of attitude control problem
- Power Systems – design and analysis of power systems
- Cost Estimation – analysis of system cost trades and impacts
- Dynamics – analysis of satellite kinetics and kinematics
- Modeling and Simulation – design, construction, and analysis of system models for optimization
- Reliability – analysis of system reliability
- Space Physics – description of system operational environment

Some of the professions involved are:

- Astronautical Engineering – design of spacecraft systems and orbital analysis
- Electrical Engineering – design of electronics
- Systems Engineering – design of complex interdisciplinary systems
- Mechanical Engineering – design of machine systems and material analysis

A.2.3.4 Scope Assessment. The scope of the system is to optimize all satellite power and attitude control functions from launch to end-of-life. We scope the problem by defining the system and the system environment.

System The system includes the energy storage and attitude control subsystems. Subsystems considered within the system scope include:

- Attitude Control Subsystem
- Power Subsystem
 - Power Management and Distribution Subsystem
 - Energy Storage Subsystem
 - Energy Generation Subsystem

System Environment The system environment consists of the satellite mission, orbital environment, and physical characteristics.

These can be simplified as:

Satellite Mission The functions which the satellite is supposed to accomplish and the parameters driven by that purpose:

1. Activity – drives power and pointing requirements
2. Duration – drives system reliability and life comparisons
3. Orbit – drives attitude control and energy storage requirements

Orbital Environment The environment in which the satellite is expected to operate:

1. Solar Intensity – will drive solar array power levels and sizing, which will affect the available power

2. Eclipse Frequency and Duration – will drive discharge times and recharge times
3. Radiation Environment – will become a life-limiting factor due to solar cell degradation and satellite electronic design
4. Atmospheric Limitations – will impact station-keeping requirements which will affect trades against momentum dumping systems, and will affect how quickly external torques saturate the flywheels
5. Magnetic Environment – will impact trades of systems employing magnetic torquers versus expendable propellant

Satellite Physical Characteristics The properties of the satellite:

1. Moments of Inertia – will directly impact flywheel sizing and rotation rate, since the flywheel will need to provide sufficient angular momentum to drive satellite orientation.
2. Mass Limits – will affect mass budget for flywheels and batteries.

A.2.3.5 Societal Sectors. We have identified the following list of societal sectors as being involved with flywheel energy storage systems.

1. Government

- (a) National Aeronautics and Space Administration (NASA)
- (b) United States Air Force (USAF)
- (c) Defense Advanced Research Projects Agency (DARPA)
- (d) Lawrence Livermore National Laboratory
- (e) Oak Ridge National Laboratory
- (f) Trinity Labs

2. Academia

- (a) Air Force Institute of Technology
- (b) University of Maryland

- (c) University of Texas
- (d) Pennsylvania State University
- (e) Texas A&M University
- (f) Auburn University
- (g) University of Virginia

3. Industry

- (a) Aerospace Corporation
- (b) World Flywheel Consortium
- (c) U.S. Flywheel Systems
- (d) SatCon
- (e) TRW
- (f) Boeing
- (g) Hughes
- (h) American Flywheel Systems
- (i) Applied Materials Technology
- (j) AFCON
- (k) FARE Inc.

A.2.3.6 Actors. The following is a list of actors involved in solving this problem:

1. 1997 System Engineering Design Project Team – Problem solution
2. AFIT/ENY Faculty – Advice and guidance
3. PL/VTV Space Vehicle Power Technologies Branch – Decision-making (customer)

A.2.3.7 Needs.

Stored Energy The energy storage subsystem must provide sufficient energy to power the satellite bus and payload during all mission phases. The energy storage subsystem must:

1. provide stored energy during the launch phase, from umbilical separation to solar array deployment
2. provide full power to the satellite bus and payload during eclipse periods
3. supplement the power provided by the solar array during peak loading periods, if necessary
4. provide power for sustainment of bus and payload during a spacecraft anomaly

Attitude Control The attitude control subsystem must provide spacecraft attitude control to meet the satellite bus and payload pointing requirements. The attitude control subsystem must:

1. meet bus and payload stability and pointing accuracy requirements
2. correct/maintain proper satellite orientation by countering external torques (within specifications)
3. reorient satellite as required by satellite bus or payload in the required time

Mission Requirements The energy storage/attitude control subsystem must meet mission requirements to include:

1. meet mass allocation and physical size constraints of satellite
2. incorporate sufficient reliability to provide power and attitude control for mission duration
3. withstand orbital environment requirements such as radiation and magnetic fields encountered

A.2.3.8 Alterables. Alterables are variables over which we have control.

They include the following:

Number of flywheels The number of flywheels used can vary greatly.

Flywheel orientation Orientation can be altered to meet attitude control needs.

Flywheel inertia Inertia encompasses material composition, size and shape of the flywheel. All can be altered to meet mission requirements, which can include energy, mass, and volume requirements.

Angular velocity Maximum angular velocity will be dependent on materials, system parameters, and reliability. Through this alterable, angular momentum capacity and energy storage capacity can be determined.

Attitude control scheme The attitude control scheme outlines how to implement attitude control on the satellite.

Battery backup Another alterable is whether or not to use a battery backup for an FES subsystem.

A.2.3.9 Constraints. Limiting boundaries of the system within which the design must stay.

Launch environment The system must be able to survive and operate despite the following rigors of launch:

Vibrations The system must tolerate vibrations induced by launch.

G-loading The system must withstand the G-force induced at launch.

Space environment The design must tolerate conditions to which it is exposed on orbit, to include:

Radiation On-orbit spacecraft are exposed to harsh radiation.

Vacuum There is no air pressure in space.

Thermal Spacecraft perform some thermal control, but there are constraints on component designs.

Vibration The design cannot allow vibrations to affect its operation.

Payload compatibility The system must meet spacecraft interface constraints, including vibration, thermal, magnetic, and physical constraints.

Cost There are limits on the maximum costs that are acceptable.

Physical limits The system may have other physical constraints.

A.3 Value System Design

A.3.1 Purpose. Once the problem is defined, the next step in the systems engineering process is to develop a method for evaluating solutions to the problem. Hall refers to this step as *Value System Design* (9). The initial task is to determine the objectives, or goals, of the project. Next, a framework for evaluating how well proposed solutions meet these objectives is established. Together, these steps combine to allow for the evaluation and comparison of possible solutions to the problem.

A.3.2 Objectives. Clear, understandable objectives outline the desired solution to the stated problem. Well-defined objectives provide the framework for the problem solution process and focus the work of the group. It is important to keep in mind that the objectives are not fixed once defined. They are subject to change if they are found to be inadequate or unnecessary. New objectives can also be introduced if the initial objectives are insufficient to fully define the desired results. The process of defining the objectives is an iterative process that continues throughout the systems engineering process.

We invested a considerable amount of time brainstorming initial objectives. The brainstorming sessions produced a number of possible objectives. We then condensed the broad, initial list of objectives to a much shorter final list. We determined that these provided the lowest level representation of the goals we set forth for the system study. We then worked up from the bottom, and grouped the lowest level objectives into more general higher-level ones. The objectives developed for our first iteration are listed below and are also shown graphically in the form of an objective tree in Figure A.2.

Optimize Satellite Design The primary objective of the project is to optimize the satellite design. This includes maximizing the satellite performance and minimizing the life cycle cost of the satellite.

Maximize Satellite Performance The model maximizes the satellite performance by maximizing the capability of the Energy Storage Subsystem (ESS) and the Attitude Control Subsystem (ACS) and by maximizing system reliability.

Maximize Energy Storage Subsystem Capability Maximize the energy storage subsystem's performance capability, focusing on the following performance areas: energy density, energy conversion efficiencies, cycle life, and depth of discharge.

Maximize Attitude Control Subsystem Capability Maximize the pointing accuracy, pointing stability and slew capability of the attitude control subsystem.

Maximize Reliability Maximize the reliability of the system design.

Minimize Life Cycle Cost Minimize the life cycle cost of the satellite by minimizing the production, launch and support costs.

Minimize Production Cost Minimize the satellite's production costs. This includes component costs as well as integration and testing costs.

Minimize Launch Cost Minimize the costs required to launch the satellite to its operational orbit.

Minimize Support Cost Minimize the cost of sustaining support required to maintain satellite health during on-orbit operations.

A.3.3 Objective Tree. The objective tree is shown in Figure A.2. This is a hierarchy of the objectives listed in Section A.3.2.

Once the objectives are established, the next step is to develop quantifiable objective measures for evaluating how well each alternative solution meets each objective. Once defined, the objective measures are also subject to revisions, if necessary.

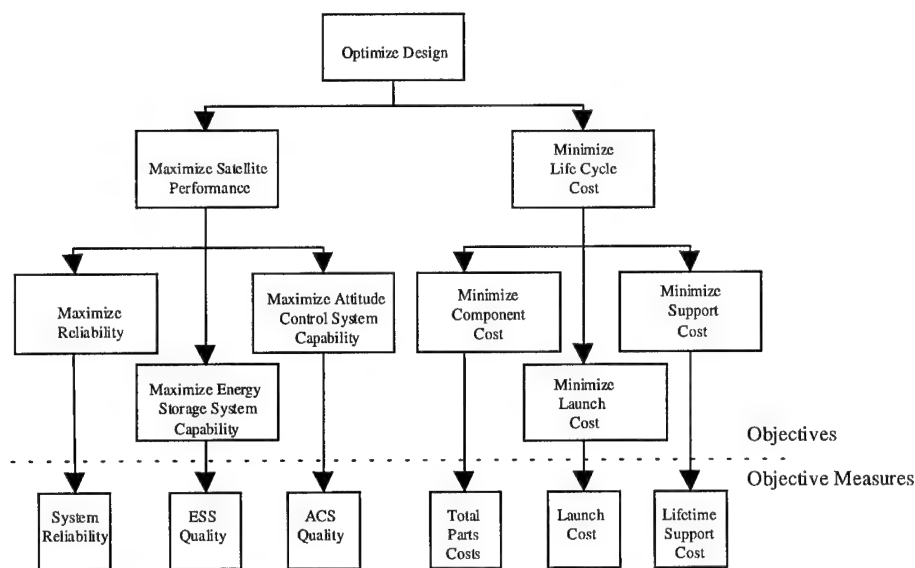


Figure A.2 Objective Tree

A.3.4 Objective Measures. The objectives measures are quantifiable values that are used to determine how well an alternative meets the above-mentioned objectives. For this iteration, we developed one objective measure for each objective. Figure A.2 shows the objective measures and their corresponding objectives. The objective measures are as follows:

Reliability The reliability of the system

ESS Quality A general measure of ESS capability, measured on 0 to 5 scale, with 5 being the highest and 0 being the lowest.

ACS Quality A general measure of ACS capability, measured on 0 to 5 scale, with 5 being the highest and 0 being the lowest.

Component Cost The cost of individual components of the system

Launch Cost The cost of the launch vehicle, launch processing, operations, and support required to put the satellite on orbit

Support Cost The cost of supporting and operating the spacecraft – this includes ground crew/engineering support personnel costs required to support satellite operations

A.3.5 Weightings. To convey the relative importance of each objective, weights are assigned to the objectives (and their corresponding objective measure). The weights for lower level branches of a particular objective (or sub-objective) sum to unity. These weights can be changed by the decision-maker to reflect his personal values. For this initial high-level analysis, we chose weights for each of the objectives based on our research and experience. These weights represented our estimate of a typical satellite program manager's values. They are listed in Table A.5.

A.3.6 Value Function. The performance of each possible solution is then analyzed using a value function. This value function converts an alternative solution's performance (in terms of how well it meets each of the objectives) to a numerical score using the above-mentioned weights. For our first evaluation, a simple linear value function was assumed and used in our analysis. The value function for this first iteration is listed below in Equation A.1.

$$\begin{aligned}
 \text{Value} = & (\text{Weight}_{\text{Reliability}})(\text{Reliability Score}) + \\
 & (\text{Weight}_{\text{ESS Quality}})(\text{ESS Quality Score}) + \\
 & (\text{Weight}_{\text{ACS Quality}})(\text{ACS Quality Score}) + \\
 & (\text{Weight}_{\text{Component Cost}})(\text{Component Cost Score}) + \\
 & (\text{Weight}_{\text{Launch Cost}})(\text{Launch Cost Score}) + \\
 & (\text{Weight}_{\text{Support Cost}})(\text{Support Cost Score})
 \end{aligned} \tag{A.1}$$

A.4 System Synthesis

A.4.1 Purpose. The purpose of the System Synthesis step is to conceptualize alternative approaches for obtaining the objectives and describe each alternative approach.

A.4.2 Process. This step was conducted at a very high level for the first iteration. The goal was to broadly define the major design alternatives and develop al-

ternative system architectures. Every architecture was included, even those that seemed unlikely to be used. This was done to help us better understand and scope the problem.

The first iteration was a quick, high-level pass over the design space with little actual analysis. We identified four alterables to be used for the high level evaluation. These were:

1. ESS Type
2. ACS Scheme
3. ESS Redundancy
4. ACS Redundancy

ESS type consisted of the three choices of using:

- Batteries only
- Flywheels only
- A combination of the two

The ACS scheme consisted of the choice of using:

- IPACS using Momentum Wheel Approach (MWA)
- IPACS using Gimbaled Flywheel Approach (GFA)
- Momentum wheels
- Control Moment Gyro (CMG)

Redundancy for both the ESS and ACS was a yes/no choice at this level. It identified that some allowance for component failure was made, but not to what degree.

These four design choices allowed for a wide range of design solutions. They ranged from traditional battery-only design with separate ACS (our baseline system for comparison purposes), to a fully integrated flywheel design that contained no batteries. All the possible combinations of these four alterables were considered. Those that were not logical or physically possible were eliminated.

A.5 Systems Analysis

A.5.1 Purpose. The purpose of the System Analysis step is to evaluate the properties of hardware performing the allocated functions.

A.5.2 Process. The first iteration was a quick, high-level pass through the systems engineering process to better understand the problem. For the first iteration, there were 36 high-level alternatives generated in the System Synthesis step. Each alternative consisted of a single hardware selection from each of the four alterables. Hence, hardware at this stage of the process was a vague term referring to categories of hardware, not specific hardware.

A.5.3 Scoring Method. We subjectively scored the general hardware categories against each other under each of the performance measures (see Table A.1). Our initial research of the literature guided the comparison and scoring. Several of the groups members worked to assign these scores. Scores were based primarily on the past experience of the members, and no attempts were made to eliminate bias. The best choice received a score of 5. The other choices were then evaluated in comparison to the best choice. If the alterable had no impact on a performance measure, then a score of zero was given to all of the hardware categories within that alterable. Under this high-level pass, performance measures aligned almost identically with objective measures. Exceptions were noted only when an alterable did not affect an objective measure. For example, ACS hardware did not affect ESS redundancy, and for ease of calculations, all were scored as zero in the performance measure. The scores of the hardware categories in each of the performance measures are listed in Table A.1.

A.6 Modeling and Optimization

A.6.1 Purpose. The purpose of this step was to model the alternatives we had generated and determine how they performed against the value system we had defined. To accomplish this goal we designed a computer program which performed the following functions:

Table A.1 First Iteration Performance Measures

| ESS Method | ACS Quality | ESS Quality | System Reliability | Launch Cost | Support Cost | Component Cost |
|---------------------------|-------------|-------------|--------------------|-------------|--------------|----------------|
| Hybrid ESS | 0 | 5 | 3 | 2 | 4 | 5 |
| Flywheels | 0 | 4 | 4 | 5 | 5 | 5 |
| Batteries | 0 | 3 | 5 | 0 | 3 | 5 |
| ESS Redundancy | | | | | | |
| Yes | 0 | 5 | 5 | 2 | 5 | 2 |
| No | 0 | 2 | 0 | 5 | 5 | 5 |
| ACS Scheme | | | | | | |
| Actuation Rate - IPACS | 5 | 0 | 3 | 5 | 5 | 4 |
| Differential Rate - IPACS | 4 | 0 | 4 | 5 | 5 | 5 |
| Separate ACS - Non-IPACS | 5 | 0 | 3 | 5 | 5 | 4 |
| ACS Redundancy | | | | | | |
| Yes | 5 | 0 | 5 | 2 | 5 | 2 |
| No | 1 | 0 | 0 | 5 | 5 | 5 |

1. Accept input from the user consisting of weights and system performance
2. Generate all feasible combinations of alterables
3. Assess the performance of the alterable combinations against the value system
4. Report the results of the analysis to the user

The functions were separated into two distinct levels at this point: a user interface and a computation engine. The front end accomplished functions 1 and 4, while the computation engine accomplished functions 2 and 3.

A.6.2 Model Concept. For this iteration we had only minimal, top-level descriptions of the functions that needed to be accomplished. We elected to perform a Quantitative Quality Function Deployment (Q2FD) analysis of these top-level objective measures to determine if we could eliminate any alternative designs early in the process. The Q2FD process drives the requirements down to the lowest level of the system and then computes the system's performance against a value system by rolling the low-level measures of system performance back up into the high-level objective measures. Rechtin and Maier provide an overview of this technique beginning on page 176 of (27). This

approach is expandable, easily implemented in a spreadsheet environment, and allows the incorporation of detailed performance models without complicating the system-level model.

A.6.3 Application and Results. The model began by determining the relative importance of each of the options in each objective measure. This was accomplished by:

- Reviewing the options
- Providing relative measures within each of the alterables and objective measures (Table A.2)
- Combining these scores to develop each alternative's score in all of the objective measures (Table A.3)
- Normalizing the scores from 0 to 1 (Table A.4)
- Finding the total score for the alternative by summing the weighted value of the normalized objective measure scores (Table A.5) Weights for the objective measures were provided by comparing the relative importance of all the objective measures. Alternative N's Total score = Score 1 * Weight 1 + ... Score n * Weight n ... + Score Z * Weight Z
- Ranking the alternatives by the total score (Table A.6)

Table A.2 Relative Importance Scheme

| <i>Alterable A</i> | <i>Objective Measure X</i> |
|--------------------|----------------------------|
| Option 1 | Score 1 |
| Option n | Score n |
| Option N | Score N |

Table A.3 Alternative Objective Scoring Scheme

| | <i>Objective Measure X</i> |
|--|-----------------------------------|
| Alternative N's option for Alterable 1 | Score A |
| Alternative N's option for Alterable n | Score n |
| Alternative N's option for Alterable Z | Score Z |
| Alternative N's score in Objective Measure X | Score 1 + Score n + ... + Score Z |

Table A.4 Normalization Scheme

| | <i>Objective Measure X</i> |
|---|--|
| Alternative 1 | Score 1 |
| Alternative N | Score N |
| Alternative Z | Score Z |
| Alternative N's normalized score in Objective Measure X | $\frac{\text{Score N} - \text{minimum}(\text{Score 1} - \text{Z})}{\text{maximum}(\text{Score 1} - \text{Z}) - \text{minimum}(\text{Score 1} - \text{Z})}$ |

A.7 Decision Making

Decision Making is the sixth step in Hall's Seven Step Process. The purpose of Decision Making is to examine the results of the optimization and modeling step and either select an alternative design for implementation or select one or more alternatives for a more detailed analysis. The latter would involve another iteration through the seven step systems engineering process.

The client is the ultimate decision maker. It is the systems engineering team's responsibility to perform the first five steps in the design process and then to present recommendations to the decision maker. The decision maker may find it valuable to see how the value system affects the outcome of the optimization step and how hypothetical changes to the value system could result in different "optimal designs."

The main purpose of this first iteration through the systems engineering process was to perform a high-level examination of the system in order to better understand and scope it. Detailed analysis was not performed during the first iteration. Consequently, the conclusions derived during the optimization step must be viewed with some trepidation. The larger benefits of this first pass are the lessons learned, the questions raised, and the resulting issues to be addressed.

Table A.5 Alternative Scoring

| | <i>Alternative N's normalized scores</i> | <i>Objective Measure weight</i> |
|---------------------|--|---------------------------------|
| Objective Measure 1 | Score 1 | Weight 1 |
| Objective Measure n | Score n | Weight n |
| Objective Measure Z | Score Z | Weight Z |

Table A.6 Top-Ten Ranked Alternatives

| <i>ESS Type</i> | <i>ESS Redundancy</i> | <i>ACS Scheme</i> | <i>ACS Redundancy</i> | <i>Score</i> |
|-----------------|-----------------------|------------------------------|-----------------------|--------------|
| Hybrid ESS | Yes | Separate | Yes | 0.80 |
| Flywheels only | Yes | Separate | Yes | 0.79 |
| Hybrid ESS | Yes | Actuation Rate with Combo | Yes | 0.79 |
| Flywheels only | Yes | Actuation Rate with Combo | Yes | 0.78 |
| Hybrid ESS | Yes | Differential Rate with Combo | Yes | 0.77 |
| Flywheels only | Yes | Differential Rate with Combo | Yes | 0.76 |
| Batteries only | Yes | Separate | Yes | 0.67 |
| Batteries only | Yes | Actuation Rate with Combo | Yes | 0.66 |
| Batteries only | Yes | Differential Rate with Combo | Yes | 0.64 |
| Hybrid ESS | Yes | Separate | No | 0.57 |
| Hybrid ESS | No | Separate | Yes | 0.57 |

With this perspective, the design team recommends the decision maker initiate a second iteration of the systems engineering process and that all alternative designs be carried forward into that second pass for a more detailed consideration and analysis.

A.8 Planning for Action - First Iteration

A.8.1 Purpose. Planning for Action is the final step in Hall's Seven Step Process. The purpose is to determine what future activities are required to advance the Systems Engineering Process (SEP) toward the desired project goals. It is necessary to identify and bring to the fore problems or incomplete areas of activity. These must be made to comply with the projected course of the project schedule. Planning for Action may also be used to identify areas which must be examined in future studies in order to better define a more favorable solution to the problem once the project is completed.

If the results of the Decision Making step suggest the implementation of a further iteration or phase of the project, or if the project is complete and there exists the need for further study, then a Planning for Action step must follow. The process is to state the desired goals and recommend future activities.

The results of the Decision Making step of the first iteration suggested that another phase of the project be initiated.

A.8.1.1 Problem Definition. In the Problem Definition step, the goals were to attain a full understanding of the problem and to define all relevant problem elements. This consisted of describing the 12 products of problem definition as they related to the SEP. There was no indication that the 12 products, as described following the first iteration, failed to clearly delineate all requisite aspects of the problem. However, there were problem elements that we decided were not needed for further investigation of the problem. Thus, in order to maintain the requisite focus of the project, activity to reassess and restate the 12 products of problem definition must be imposed.

A.8.1.2 Value System Design. The goals of the Value System Design step were to define the objectives of the system study, define the objective measures of the stated objectives, and assign subjective weights to the stated objective measures. The objectives were well-defined following the first iteration. However, the objective measures were defined at a high level and require further delineation. Also, information that could be used to develop weights that the team could apply with confidence was not yet available. The objective measures and weights had to be well-developed in order for any system model created to produce viable data for analysis. This circumstance was one factor making it impossible to perform any detailed analysis in the early phase of the project. Thus, it is necessary to focus activity toward further defining the objective measures and weights.

A.8.1.3 System Synthesis. In the System Synthesis step the goals were to conceptualize alternative approaches for obtaining the objectives and to describe each alternative approach. The first iteration was a broad examination and the subsequent alternatives were not described in detail at this level. This was another factor affecting the efficacy of formulating a system model with which we could perform detailed analysis of the problem. Thus, it is important that activity be directed toward gaining a further breakdown of the alternatives.

A.8.1.4 System Analysis. The goal of the System Analysis step was to evaluate the properties of the hardware elements performing the allocated functions that determine the performance measures. Some progress was made regarding the formulation

of a scoring method based on the measures. This is necessary in order to determine the degree to which the alternatives satisfy the design needs. With a viable scoring method in hand, we can model alternatives and compare them in order to state which one is the optimum configuration. Given the high-level approach to the first iteration, and the slight extent to which the alternatives were defined, this step could produce only superficial results. System Analysis should proceed after the activities related to prior steps have been addressed in the next iteration of the SEP.

A.8.1.5 Modeling and Optimization. In the Modeling and Optimization step the goal was to model the alternatives that were generated in the System Synthesis step and determine how they perform with respect to the value system we have defined. For the first iteration, this step consisted of a general description of the model's form and configuration. Having been left with high-level results, we decided on the Q2FD analysis to determine whether or not some alternatives may be eliminated early in the process. Once a more detailed breakdown of the system is achieved in the next iteration, activity in this step can proceed to defining the final system model.

A.8.1.6 Systems Engineering Process. Hall's model provides an initial starting point, but is too general in its bare form. Prior to starting activity on the second pass, we must step back and examine how we can customize Hall's approach to our specific problem.

Appendix B. Second Iteration

B.1 Lessons Learned from First Iteration

B.1.1 Purpose. McNabb (20) defines *Lessons Learned* as, "Knowledge acquired by individuals or groups, regarding positive or negative aspects of efforts performed by themselves or others." *Lessons Learned*, as one may guess, is an activity that is closely associated with record keeping. An individual or group is assigned to the task. As the project progresses, the activities associated with partitioned areas of interest are continually scrutinized. Lessons learned as a result of the observation of the activities are recorded for future reference. Should these areas of interest be repeated in future iterations of the process, the lessons learned will form a valuable tool for optimizing the associated time and effort required to complete the activities.

Since the design study is based on Hall's Seven Step Process (which is an iterative endeavor), it is requisite to identify such positive and negative aspects in the different areas of activity. In so doing, we can avoid the repetition of mistakes, and perform the activities in a way that maximizes the utility of the time and effort.

B.1.2 Results. The following are the *Lessons Learned* from the first iteration:

- Make the acquisition of input data a priority early in the process. This helps to avoid schedule delays brought about by slow or uncooperative sources, and/or having to settle for decreased performance due to an insufficient database.
- Understand the architecture of the software model and the hierarchy of interactions among the modules and the main program. This will aid management in determining communications channels among the individual programmers and will allow for a more informed approach to planning and organizing the development effort.
- Assign activities such as keeping a record of important decisions and lessons learned at the project's inception. This will help avoid divining the requisite information in retrospect.

- In the software development effort, management must be flexible. The optimum course of action is to allow those individuals with background knowledge and experience in the software technology to take the lead. Thus, delays and performance degradation resulting from a lack of understanding are more likely to be avoided. Love (19) writes, "Managing software projects requires a detailed technical understanding of the software management business and the software technology itself."
- Time constraints require a model that can rapidly evolve, adapt, and grow as the understanding of the problem develops. It is not possible to run through the entirety of Hall's Method each time one considers changing the basic approach. *Integrated Modeling* is a must. The internal iterative engine it provides allowed us to rapidly deploy changes to the lowest level of our analysis.

B.2 Problem Definition

B.2.1 Purpose. The purpose of the Problem Definition step is to clearly identify the problem being solved; the system being designed; the people and organizations involved; and the needs, alterables, and constraints affecting the overall design.

Although our understanding of the problem increased dramatically during the course of the design study, the formal Problem Definition, as described in Appendix A, did not change significantly. The only notable edit is within the *Alterables* portion of our Problem Definition.

The entire Problem Definition is listed again below, with the change in the *Alterables* incorporated.

B.2.2 The Products of Problem Definition. Sage (29) describes several products of Problem Definition. Nine of them are listed below. During the second iteration of the Systems Engineering Process (SEP), no additional products were added to the list nor were any deleted.

1. A well conceived title for the problem

2. A descriptive scenario explaining the nature of the problem and how it came to be a problem, presenting as much history as possible with the available resources
3. An understanding of the disciplines and professions relevant to the problem
4. An assessment of scope
5. A determination of the societal sectors involved
6. An identification of the actors involved in the problem solving situation
7. An identification of *needs*
8. An identification of *alterables*
9. An identification of major *constraints*

We contrived these products of Problem Definition for our system study. They are identified and described below.

B.2.2.1 Title. Satellite Integrated Power and Attitude Control System Design Study

B.2.2.2 Scenario. Current Earth-orbiting satellites use solar arrays for primary power and chemical batteries to store energy for use during eclipse periods. Satellite load demands for electrical power are mission dependent. Batteries are required to supply electrical power during periods of spacecraft eclipse for a certain number of charge/discharge cycles over the satellite's lifetime. The frequency and duration of the eclipses a mission encounters varies with orbital altitude and inclination. Typical eclipse seasons for geostationary orbits last for 45 days and occur twice a year. The maximum eclipse duration for geostationary satellite orbits is 72 minutes. Eclipse seasons for LEO satellites are much more dependent on altitude and inclination. They can vary from two weeks per year up to the entire year for some satellites. For a typical 100 minute LEO period, the maximum eclipse duration is 36 minutes.

Depth-of-discharge refers to the percentage of total battery capacity that is removed during a discharge period. For traditional aerospace chemical batteries, battery cycle life decreases with increasing depth-of-discharge. A longer satellite lifetime requirement

dictates that a lower depth-of-discharge be used for the batteries. Therefore, to meet energy storage requirements over the satellite lifetime, batteries are oversized. Thus, there is an imposition of additional mass with no proportionate benefit.

A Flywheel Energy Storage (FES) system does not suffer from decreasing cycle life with higher depths-of-discharge. An FES system stores energy mechanically in the form of a rotating flywheel. An FES system does not need to be oversized. The excess mass and inaccessible capacity associated with chemical batteries can be avoided. These flywheels can be designed with a higher specific energy than batteries. An FES system may have less mass than chemical batteries while satisfying the same energy storage requirement.

Devices incorporating rotating wheels are used extensively on-board satellites in attitude control systems to absorb or condition satellite angular momentum. Attitude control is accomplished either by differential rotation of a flywheel array, or by applying external torques to the flywheels through an actuator and gimbal assembly. Rotational speeds on the order of 2,000 to 4,000 RPM are common. When the wheels become "saturated," they have reached the specified limit of their rotational speed. At this point, it becomes necessary to apply external torques to transfer the excess momentum out of the system. This is accomplished by expending propellant through the reaction control system, or by using a magnetic or gravitational torquer (where feasible). Many satellites reach their end-of-life when they have exhausted their propellant supply and can no longer maintain the required attitude.

The flywheels used in an FES system can replace the reaction and momentum wheels used for attitude control. The resulting system is called an Integrated Power and Attitude Control System (IPACS). By achieving higher angular speeds than traditional reaction and momentum wheels, flywheels can be made with a larger angular momentum storage capacity. This may lead to less frequent use of the reaction control system to diminish accumulated momentum. Therefore, a reduction in propellant use could extend the life of the satellite.

The problem of determining the characteristics of a combined electrical power system and ACS architecture that optimizes the mission requirements is addressed in this study.

The analysis is performed on the following architectures:

- separate power and ACS using batteries for energy storage and traditional ACS
- separate power and ACS using flywheels for energy storage and traditional ACS
- flywheel IPACS

B.2.2.3 Relevant Disciplines/Professions. Some of the disciplines involved are:

- Attitude Control – design and analysis of attitude control problem
- Power Systems – design and analysis of power systems
- Cost Estimation – analysis of system cost trades and impacts
- Dynamics – analysis of satellite kinetics and kinematics
- Modeling and Simulation – design, construction, and analysis of system models for optimization
- Reliability – analysis of system reliability
- Space Physics – description of system operational environment

Some of the professions involved are:

- Astronautical Engineering – design of spacecraft systems and orbital analysis
- Electrical Engineering – design of electronics
- Systems Engineering – design of complex interdisciplinary systems
- Mechanical Engineering – design of machine systems and material analysis

B.2.2.4 Scope Assessment. The scope of the system is to optimize all satellite power and attitude control functions from launch to end-of-life. We scope the problem by defining the system and the system environment.

System The system includes a satellite's power system and attitude determination and control system. Subsystems considered within the system scope include:

- Attitude Determination and Control System
 - Attitude Control Subsystem (ACS)
- Power Subsystem
 - Power Management and Distribution Subsystem (PMAD)
 - Energy Storage Subsystem (ESS)
 - Energy Generation Subsystem (EGS)

System Environment The system environment consists of the satellite mission, orbital environment, and physical characteristics.

These can be simplified as:

Satellite Mission The functions which the satellite is supposed to accomplish and the parameters driven by that purpose:

1. Activity – drives power and pointing requirements
2. Duration – drives system reliability and life comparisons
3. Orbit – drives attitude control and energy storage requirements

Orbital Environment The environment in which the satellite is expected to operate:

1. Solar Intensity – will drive solar array power levels and sizing, which will affect the available power
2. Eclipse Frequency and Duration – will drive discharge times and recharge times
3. Radiation Environment – will become a life-limiting factor due to solar cell degradation and satellite electronic design
4. Atmospheric Limitations – will impact station-keeping requirements which will affect trades against momentum dumping systems, and will affect how quickly external torques saturate the flywheels

Satellite Physical Characteristics The properties of the satellite:

1. Moments of Inertia – will directly impact flywheel sizing and rotation rate, since the flywheel will need to provide sufficient angular momentum to drive satellite orientation.
2. Mass Limits – will affect mass budget for flywheels and batteries.

B.2.2.5 Societal Sectors. We have identified the following list of societal sectors as being involved with flywheel energy storage systems. In the course of the design study, we interacted with representatives from almost all of these organizations.

1. Government

- (a) National Aeronautics and Space Administration (NASA)
- (b) United States Air Force (USAF)
- (c) Defense Advanced Research Projects Agency (DARPA)
- (d) Lawrence Livermore National Laboratory
- (e) Oak Ridge National Laboratory
- (f) Trinity Labs

2. Academia

- (a) Air Force Institute of Technology
- (b) University of Maryland
- (c) University of Texas
- (d) Pennsylvania State University
- (e) Texas A&M University
- (f) Auburn University
- (g) University of Virginia

3. Industry

- (a) Aerospace Corporation

- (b) World Flywheel Consortium
- (c) U.S. Flywheel Systems
- (d) SatCon
- (e) TRW
- (f) Boeing
- (g) Hughes
- (h) American Flywheel Systems
- (i) Applied Materials Technology
- (j) AFCON
- (k) FARE, Inc.

B.2.2.6 Actors. The following is a list of actors involved in solving this problem:

1. PL/VTV Space Vehicle Power Technologies Branch (sponsor/customer) – Decision-making
2. 1997 System Engineering Design Project Team – Problem solution
3. AFIT/ENY Faculty – Advice and guidance

B.2.2.7 Needs.

Stored Energy The energy storage subsystem must provide sufficient energy to power the satellite bus and payload during all mission phases. The energy storage subsystem must:

1. Provide stored energy during the launch phase, from umbilical separation to solar array deployment
2. Provide full power to the satellite bus and payload during eclipse periods
3. Supplement the power provided by the solar array during peak loading periods

4. Provide power for sustainment of bus and payload during a spacecraft anomaly

Attitude Control The attitude control subsystem must provide spacecraft attitude control to meet the satellite bus and payload pointing requirements. The attitude control subsystem must:

1. Meet bus and payload stability and pointing accuracy requirements
2. Correct/maintain proper satellite orientation by countering external torques (within specifications)
3. Reorient satellite as required by satellite bus or payload in the required time

Mission Requirements The energy storage/attitude control subsystem must meet mission requirements to include:

1. Meet mass allocation and physical size constraints of satellite
2. Incorporate sufficient reliability to provide power and attitude control for mission duration
3. Withstand orbital environment requirements such as radiation and magnetic fields encountered

B.2.2.8 Alterables. Alterables are variables over which we have control, and which we vary in order to optimize the design.

The Alterables received considerable rework from the first iteration to the second iteration. We updated the alterables to accurately reflect our modeling. The new alterables are as follows:

Number of flywheels and their orientation The number of flywheels used can vary greatly, and their orientation can be altered to meet attitude control and energy storage needs.

Number of batteries The number of batteries used can also be varied to meet energy storage requirements.

ESS Type ESS can be accomplished using either flywheels or chemical batteries.

ACS Scheme ACS can be accomplished one of three different ways:

- A traditional attitude control system
- A gimbaled flywheel array
- A differential rate flywheel array

Launch Configuration of Flywheels The flywheels can either be locked or spinning at launch.

PMAD Approach The PMAD system can either be a Direct Energy Transfer system or a Peak Power Tracking system.

B.2.2.9 Constraints. Limiting boundaries of the system within which the design must stay.

Launch environment The system must be able to survive and operate despite the following rigors of launch:

Vibrations The system must tolerate vibrations induced by launch.

G-loading The system must withstand the G-force induced at launch.

Space environment The design must tolerate conditions to which it is exposed on orbit, to include:

Radiation On-orbit spacecraft are exposed to harsh radiation.

Vacuum There is no air pressure in space.

Thermal Spacecraft perform some thermal control, but there are constraints on component designs.

Vibration The design cannot allow vibrations to affect its operation.

Payload compatibility The system must meet spacecraft interface constraints, including vibration, thermal, magnetic, and physical constraints.

Cost There are limits on the maximum costs that are acceptable.

Physical limits The system may have other physical constraints.

B.3 Value System Design

B.3.1 Purpose. The purpose of *Value System Design* is to develop a method for evaluating solutions to the problem by determining the objectives of the project and constructing a framework for measuring how well proposed solutions meet these objectives.

B.3.2 Objectives. The iterative nature of the systems engineering process allowed the team to update any portion of the model during the system study. The results of our initial analysis indicated the need for quantifiable additional detail in the model.

As we expanded the level of detail in the model, we determined that our initial objectives also required a second look. Our on-going research, coupled with the experience gained from the first iteration, highlighted the shortcomings of the initial objectives. Some of the objectives were over-specified initially, while other key objectives were omitted. Additionally, the initial objectives were poorly grouped. The remainder of this section details the changes made to the study objectives, and includes justifications for the changes we made.

The initial objectives from the first iteration are listed below as a reminder:

- Optimize Satellite Design
 - Maximize Satellite Performance
 - * Maximize Energy Storage Subsystem Capacity
 - * Maximize Attitude Control Subsystem Capability
 - * Maximize Reliability
 - Minimize Life Cycle Cost
 - * Minimize Production Cost
 - * Minimize Launch Cost
 - * Minimize Support Cost

In contrast to the objectives brainstorming session we held during the first iteration, we redefined the study's objectives during the modeling portion of the second iteration. The objectives evolved during the second iteration primarily in response to our increased

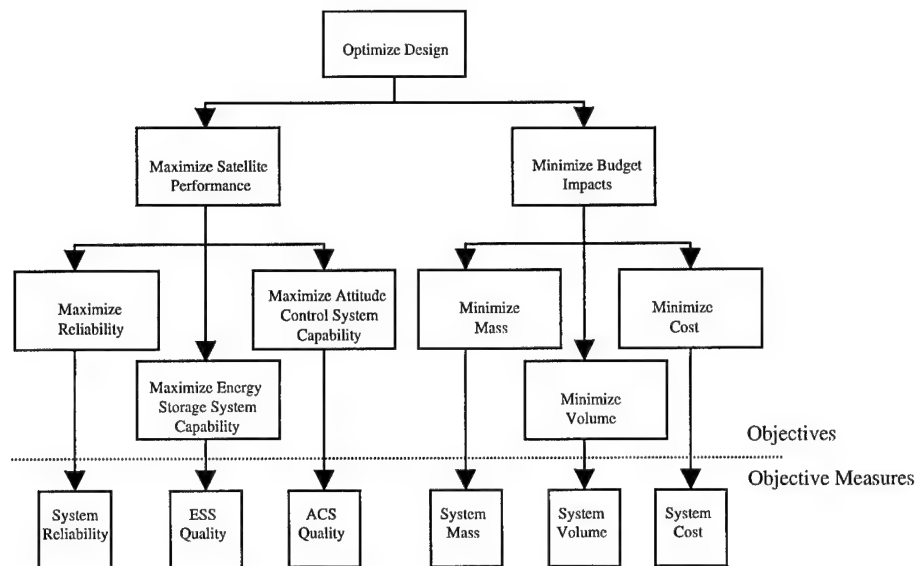


Figure B.1 Objective Tree for Second Iteration

knowledge of the subject areas. They also changed as a result of modeling decisions. The changes to the objectives and the reasons for the changes are documented below. A list of the final objectives along with an updated objective tree is also presented.

The primary objective, *Optimize Design*, did not change. However, how we accomplished this objective received considerable attention during the second iteration rework of the objectives. We modified the major sub-objectives one level below the overall goal of optimizing the design (see Figure B.1). The redefining of the Energy Storage Subsystem objective (see below) primarily forced this change. The new objectives represented distinct goals for the study, and needed to stand alone.

As mentioned above, the *Maximize Energy Storage Subsystem Capability* objective was redefined to present a more realistic objective of the energy storage function on-board a spacecraft. We made this change (in conjunction with several other changes to the cost objectives) to reflect a more realistic scenario for a satellite designer. The revised goal is to meet the energy storage requirement for the mission.

Satellite designers not only desire solutions that meet their needs for performance, but also fit within cost or budget constraints. For both energy storage systems and at-

titude control systems, any hardware combination that meets the missions needs will be considered. However, the most efficient (or best) hardware set will be one that meets the needs with the least mass, volume, and cost. Therefore, we identified minimizing mass and volume as new, independent objectives.

We then reviewed our initial cost objectives and concluded that the costs of a particular solution were incorrectly addressed by the first iteration objectives. The initial cost objectives included production, launch and support costs. Two of these costs objectives were eliminated to match modeling decisions made during the second iteration.

The launch costs were one of the objectives eliminated. Launch costs are a function of mass, so to score both launch costs and mass separately would be doubling the emphasis on mass. Also, the modeling approach we took only addresses the particular spacecraft systems of interest to our study, the attitude control subsystem and power system. Launch costs are based on the mass and volume of an entire spacecraft, not just specific portions of a spacecraft. Therefore, we could not accurately develop launch costs for our system, because our system boundary prohibited it.

The other change eliminated the *Minimize Support Cost* objective. This objective was initially introduced to capture the time and cost of on-orbit chemical battery reconditioning. However, research into battery reconditioning indicated that the current generation and future generations of chemical batteries are not subject to periodic reconditioning. Based on that information, we decided the objective was no longer relevant, and dropped it from our list.

The last change reorganized the objectives and collected ones with similar goals. Since mass, volume, and cost are all budget items, the *Minimize Mass*, *Minimize Volume*, and *Minimize Cost* objectives were grouped together under a higher level objective called *Minimize Budget Impacts*.

The final system study objectives are as follows:

Optimize Design Optimize the PACS design to maximize its performance and minimize its life cycle cost.

Maximize Satellite Performance Maximize the capability of the power and attitude control subsystems.

Maximize Reliability Maximize the reliability of the PACS components considered.

Maximize Energy Storage System Capability Maximize the ability of the energy storage subsystem to store and discharge energy.

Maximize Attitude Control System Capability Maximize the torque capability and momentum storage capacity of the attitude control subsystem.

Minimize Budget Impacts Minimize the budget impacts of the PACS.

Minimize PACS Mass Minimize the mass of the PACS components.

Minimize PACS Volume Minimize the volume of the PACS subsystems.

Minimize PACS Costs Minimize the non-recurring and recurring production costs of the PACS components.

B.3.3 Objective Measures. Revised objectives also drove changes to the objective measures. We redefined our objective measures to match the new objectives. Additionally, the new objectives measures were grouped into three broad categories:

- Reliability
- Performance
- Budget

For reliability, the objective measure is the overall PACS reliability. The performance objective measures are:

- Excess Energy Storage Capacity
- Torque Capability
- Excess Momentum Storage Capacity

Finally, the budget objective measures include system mass, system volume, and the two cost values (non-recurring and recurring). Our goal is to minimize use of these budget resources.

The final list of performance measures is:

Reliability The reliability of the system.

Energy Storage Subsystem Quality The amount of excess storage capacity in the ESS.

Attitude Control System Component Quality A measure of ACS capability based on a combination of torque capability and momentum storage capacity.

Mass The mass of the ESS, ACS, PMAD, and EGS systems.

Volume The volume of the ESS, ACS, PMAD, and EGS systems.

Cost The sum of the non-recurring and recurring costs of the system.

B.3.4 Value System Weighting. The process for assigning weights did not change from the first iteration to the second iteration. The second iteration's weights were not fixed, as they were in the first iteration. The weights varied and depended on the particular analysis being conducted.

B.3.5 Value Function. The performance of each alternative solution was still analyzed using a value function. The simple linear function value function was retained, but altered slightly for the second iteration to reflect the changes made to the objectives and the objective measures. The implementation details of the value function are explained in further detail later in *Modeling and Optimization*. The final value function we developed is listed below in Equation B.1.

$$Value_{system} = \bar{W}^T \bar{X} \quad (B.1)$$

$$\bar{W} = \begin{pmatrix} ACS \text{ Quality Weight} \\ ESS \text{ Quality Weight} \\ Cost \text{ Weight} \\ Mass \text{ Weight} \\ Volume \text{ Weight} \\ Reliability \text{ Weight} \end{pmatrix} \quad (B.2)$$

$$\bar{X} = \begin{pmatrix} ACS \text{ Quality} \\ ESS \text{ Quality} \\ Normalized \text{ Cost} \\ Normalized \text{ Mass} \\ Normalized \text{ Volume} \\ Normalized \text{ Reliability} \end{pmatrix} \quad (B.3)$$

B.4 System Synthesis

B.4.1 Purpose. The purpose of the System Synthesis step (29) is to:

- Conceptualize alternative approaches for obtaining the objectives
- Describe each alternative approach

B.4.2 Process. This step was conducted at a high level for the second iteration. The goal was to broadly define the major design alternatives and develop all possible alternative system architectures.

We identified four alterables to be used for the evaluation. These were:

1. ESS Type
2. ACS Scheme
3. Flywheels spinning or locked during launch, and
4. PMAD Type

ESS type consisted of the choice of using:

- Batteries only, or
- Flywheels only

The ACS scheme consisted of the choice of using:

- IPACS using MWA
- IPACS using GFA
- Momentum wheels
- CMG

The alterable of locking the flywheels during launch or spinning them only applied to flywheel-based designs. The locked option included a primary battery to provide power during launch. The alterable of PMAD type included either a Direct Energy Transfer system or a Peak Power Tracking system.

These four design choices allow for a wide range of design solutions. They ranged from traditional battery-only design with separate ACS (our baseline system for comparison purposes), to a fully integrated flywheel-only design (no batteries). All the possible combinations of these four elements were considered—those that were not possible were eliminated.

B.5 Systems Analysis

B.5.1 Purpose. The purpose of the System Analysis step is to analyze the proposed alternatives generated during System Synthesis.

B.5.2 Process. The simple method used during the first pass did not provide the level of detail to differentiate between alternatives. Therefore, we developed an entirely new method to analyze the alternatives during the second iteration. We developed this method concurrently with our changes to the objectives and objective measures. We chose to model the lower level functions required to meet our objectives with equations. We collect and score combinations of alterables that meet all of the mission requirements. Then, we compare the performance of these functions to our objective measures. This

section details the development of the equations we use to model our functions and compute performance measures. These performance measures are then compared to our objective measures in the system model, which is described in the following section.

B.5.2.1 Assumptions. To effectively model the different alternatives, we made the following assumptions:

- The spacecraft is in a circular orbit.
- The worst-case external torque is a single torque equal to the sum of all the external torques acting perpendicular to the spacecraft's angular momentum vector.
- External torques can be generalized over an orbit into secular or cyclical
- The ACS components that can be varied are the flywheels, the addition of a gimbal unit, and the addition of a separate set of attitude control reaction wheels or control moment gyros. Components, such as electronics, thrusters, and sensors that are common in all ACS alternatives are ignored for calculation of performance measures. Components that are normally part of the ESS are not included in the ACS module for calculation of mass, size, and volume.
- All flywheels used in a single alternative are identical.
- The flywheels have an upper- and lower-rated rotation rate.
- Flywheels experience no loss of performance/capacity over time.
- Efficiencies for the motor and generator remain constant.
- Power loads for the magnetic bearings and controller remain constant.
- Other small losses and loads are ignored.
- Battery and satellite bus voltages are nominally 28 V.
- Lithium Ion (Li-Ion) batteries are only applicable for missions requiring fewer than 2000 charge/discharge cycles due to current Li-Ion cycle life limitations.
- The spacecraft bus is fully regulated.

- The wiring efficiencies are 99% between the solar arrays and PMAD components, and 99.5% between all other components.
- The power distribution unit efficiency is the same for all architectures.
- The PPT efficiency is the same for all architectures that incorporate it.
- The end-of-life PMAD system reliability is 99%.
- The model assumes that solar arrays provide the energy generation function.
- The temperature of the solar array decreases linearly as the altitude of the spacecraft increases from LEO to GEO.

B.5.3 Inputs. The following inputs are used to create the required output and performance measures.

Table B.1 Inputs - Hardware Data

| Name | Units | Description | Functional Area (FA) |
|------------------------|-------------------|--|----------------------|
| P_{Wheel} | W | Flywheel housekeeping power | PMAD |
| T_{taca} | Nm | The peak torque that the separate ACS is capable of providing the spacecraft | ACS |
| P_{taca} | W | The power necessary to operate the separate ACS | ACS |
| $ArraySP$ | W/kg | Specific power of selected array material | EGS |
| $AnnDegsA$ | | Annual degradation of selected array material | EGS |
| $MaxAngInc$ | deg | Maximum angle of incidence on solar array | EGS |
| I_s | kg m ² | Inertia of the rotor and any components spinning with it | ESS ACS |
| ω_{max} | rad/s | Maximum operating rotational speed of the rotor | ESS ACS |
| ω_{min} | rad/s | Minimum operating rotational speed of the rotor | ESS ACS |
| $Controller Power$ | W | Average power consumption of flywheel controller | ESS |
| $Bearing Power$ | W | Average power consumption of the magnetic bearings | ESS |
| $FlyMass$ | kg | The mass of one flywheel | ESS |
| $FlyVolume$ | m ³ | The volume of one flywheel | ESS ACS |
| $FlyMass_{LR}$ | kg | The additional mass incurred in a launch-rated fly-wheel | ESS |
| $FlyVol_{LR}$ | m ³ | The volume delta for a flywheel rated for launch | ESS |
| $Flywheel Reliability$ | | Probability a flywheel will be functional at end-of-life | ESS |
| $Battery Type$ | | Type of battery (Ni-Cd, Ni-H ₂ , or Li-Ion) | ESS |
| $Rated Capacity$ | Ahr | Rated capacity of each battery | ESS |
| $Battery Mass$ | kg | Unit mass of each battery | ESS |
| $Battery Volume$ | m ³ | Unit volume of each battery | ESS |
| T_{mg} | Nm | The peak torque that the motor is capable of delivering to the rotor | ACS |
| Res | % | The accuracy to which the wheel speed can be controlled | ACS |

Table B.2 Calculations Passed Between Hardware Models

| Name | Units | Description | FA |
|-----------------|-------|--|-------------|
| E_{Slew} | Whr | Energy for slew | PMAD |
| E_{GEO} | Whr | ACS GEO Orbit Energy | PMAD |
| P_{ACS} | W | ACS operating power | PMAD |
| P_{SA} | W | Power provided by the solar arrays at the end-of-life | ACS |
| $P_{EG,BOL}$ | W | Total solar array output power at beginning-of-life (BOL) for a Flywheel ESS | PMAD |
| $P_{ES,BOL}$ | W | The portion of solar array output power required to store energy at BOL | PMAD |
| P_{OpBatt} | W | Operating power PMAD needs from EGS for loads at end-of-life (EOL), given batteries | EGS |
| $P_{RpEOLBatt}$ | W | Recharge power PMAD needs from EGS to recharge batteries at EOL | EGS |
| $P_{TotEOLFly}$ | W | Total power PMAD needs from EGS at EOL, given flywheels | EGS |
| T_{Orb} | min | The time it takes the satellite to complete one orbit | PMAD ACS |
| T_{MaxEcl} | min | Maximum amount of time that the satellite is in darkness | PMAD |
| Ecl_{max} | | | ACS |
| E_{MaxEcl} | Whr | The total energy (integrated power) required at the load input over the maximum eclipse duration | PMAD |
| N_{Batt} | | The number of batteries needed to satisfy the energy storage requirement | PMAD |
| N_{Fly} | | The number of flywheels needed to satisfy the energy storage and attitude control requirements | PMAD |
| N | | | ACS |

Table B.3 Inputs - Efficiencies

| Name | Units | Description | FA |
|-----------------|-------------|---|------|
| η_{PPT} | | PPT efficiency which accounts for the PPT accuracy and series regulation losses | PMAD |
| η_{PDU} | | PDU efficiency which accounts for losses associated with power flow through the PDU | PMAD |
| η_{BCC} | | Battery charge controller (BCC) efficiency which accounts for losses associated with power flow through the BCC during battery charging | PMAD |
| η_{BDC} | | Battery discharge controller (BDC) efficiency which accounts for losses associated with power flow through the BDC during battery discharge | PMAD |
| η_{BC} | | Battery charge efficiency which accounts for the losses associated with the battery's ability to store energy | PMAD |
| η_{BD} | | Battery discharge efficiency which accounts for the losses associated with the battery's ability to provide energy | PMAD |
| $ArrayEff$ | | Efficiency of selected array material | EGS |
| η_{TempSA} | deg C -1 | Temp inefficiency of selected array material (per deg C above 28 deg C) | EGS |
| η_{PCM} | | Power converter for motoring (PCM) efficiency which accounts for losses associated with power flow through the power converter that conditions the input power to the flywheel motor | PMAD |
| η_{PCG} | | Power converter for generating (PCG) efficiency which accounts for losses associated with power flow through the power converter that conditions the output power from the flywheel generator | PMAD |
| η_{Mot} | | Motor efficiency which accounts for the losses associated with the flywheel motor's ability to store energy in the flywheel | PMAD |
| η_{Gen} | | Generator efficiency which accounts for the losses associated with the flywheel generator's ability to provide energy | PMAD |
| η_{EGS-ES} | | Energy generation to energy storage (flywheel) efficiency | ACS |
| η_{DesSA} | | Efficiency associated with design of solar array | EGS |
| η_{ES-L} | | Energy storage (flywheel) to loads efficiency | ACS |
| η_{EGS-L} | | Energy generation to loads efficiency | ACS |
| η_{IW} | | Flywheel to flywheel efficiency | ACS |

Table B.4 Inputs - Mission Data

| Name | Units | Description | FA |
|--|-------------------|---|----------------|
| <i>Design Life</i> | years | Mission design life requirement | EGS ESS ACS |
| <i>M_{Dry}</i> | kg | The mass of the payload and spacecraft systems (not including propellant mass) | PMAD |
| <i>I_x, I_y, I_z</i> | kg m ² | The principal moments of inertia of the spacecraft | ACS |
| <i>A</i> | m ² | The maximum cross sectional area of the spacecraft | ACS |
| <i>ArrayCfg</i> | | Array configuration (Panel-mounted, body-mounted/ cylindrical, or body-mounted/ omni) | EGS |
| <i>PMAD_{Type}</i> | | Determines whether to model a DET based system or a PPT based system | PMAD |
| <i>t_{slew}</i> | minutes | The minimum expected duration for the slew maneuver | ACS |
| <i>N_{slews}</i> | | The expected number of slew maneuvers per orbit | ACS |
| <i>θ_{slew}</i> | degrees | The maximum expected angular displacement of a slew maneuver | ACS |
| <i>P_n</i> | W | The power requirements of the loads over the orbit, broken up into a maximum of 5 discrete power levels and durations | PMAD ACS |
| <i>LD_n</i> | minutes | The duration of the associated power requirement | ACS |
| <i>P_{Min}</i> | W | The lowest non-zero load from the load profile | PMAD |
| <i>Altitude</i> | km | Altitude for selected mission | EGS ESS ACS |
| <i>θ_{maxdev}</i> | degrees | Maximum deviation from Earth-pointing allowed | ACS |

B.5.3.1 Constants and Assumed Values.

The following values are either constants or assumed values used in the calculations.

Table B.5 Constants

| Name | Value | Description | FA |
|--------------------|---|--|-----|
| C_d | 4 | Drag coefficient | ACS |
| $c_{pa} - c_g$ | 0.1 m | Distance between the center of atmospheric pressure and the center of gravity of the spacecraft | ACS |
| $c_{ps} - c_g$ | 0.1 m | Distance between the center of solar pressure and the center of gravity of the spacecraft | ACS |
| μ | $398601 \text{ km}^3/\text{s}^2$ | Gravitational constant for Earth | ACS |
| D | 1 amp-turn-m^2 | Residual magnetic dipole of the spacecraft | ACS |
| M | $7.96 \times 10^{15} \text{ m}^3 \text{ Tesla}$ | Magnetic moment of the Earth | ACS |
| F_i | 2 | Inclination factor accounts for the magnetic field changing with respect to the magnetic inclination | ACS |
| K | 1358 W/m^2 | Average intensity of the solar radiation at Earth | ACS |
| $MinSInt$ | 1309 W/m^2 | Minimum solar intensity at aphelion | EGS |
| q | 0.6 | Reflectivity of the spacecraft | ACS |
| c | $3 \times 10^8 \text{ m/s}$ | Speed of light | ACS |
| <i>Bus Voltage</i> | 28 V | Bus voltage | ESS |

B.5.3.2 Hardware Model Outputs. The functional areas create the following outputs.

Table B.6 Outputs

| Name | Units | Description | FA |
|----------------------|----------------|--|---------------------|
| E_{Req} | Whr | The total amount of energy that must be stored while the satellite is in sunlight, given a battery ESS | PMAD |
| $P_{ES,EOL}$ | W | The portion of solar array output power required to store energy in the batteries at EOL | PMAD |
| P_{Op} | W | The portion of solar array output power required to support the loads, given a battery ESS | PMAD |
| E_{Req} | Whr | The total amount of energy that must be stored in the flywheels while the satellite is in sunlight | PMAD |
| $P_{EG,EOL}$ | W | Total solar array output power at EOL for a Flywheel ESS | PMAD |
| η_{Fly} | | The total efficiency for power transfer between flywheels | PMAD |
| η_{EG-ES} | | Total efficiency for power transfer from the solar array output to the flywheel input | PMAD |
| η_{EG-L} | | Total efficiency for power transfer from the solar array output to the load input | PMAD |
| $FlyCapacity$ | Whr | The energy storage capacity of a flywheel | ESS |
| $FlyPower$ | W | The average housekeeping power required by a flywheel | ESS |
| $Effective Capacity$ | Whr | The effective energy storage capacity of the battery | ESS |
| $Torque Capability$ | Nm | Torque capability beyond the minimum requirements | ACS |
| $Momentum Storage$ | Nms | Momentum storage beyond the minimum requirements | ACS |
| $Power Requirement$ | W | Peak power required by the ACS system | ACS |
| $Mass$ | kg | Total mass of the subsystem | ESS PMAD EGS ACS |
| $Volume$ | m ³ | Total volume of the subsystem | ESS PMAD EGS ACS |
| $Recurring Cost$ | \$K | Cost to produce one unit of the subsystem | ESS PMAD EGS ACS |
| $Non-recurring Cost$ | \$K | Cost to research and develop the subsystem | ESS PMAD EGS ACS |
| $Reliability$ | | Reliability of the subsystem | ESS PMAD EGS ACS |
| $ACS Eclipse Energy$ | Whr | Energy requirement to operate ACS during eclipse | ACS |
| E_{ACS} | Whr | Energy requirement to operate ACS over a single orbit | ACS |
| $ACS Flywheels$ | | Minimum number of flywheels necessary for ACS | ACS |

B.5.4 Energy Storage Unit Sizing—Flywheels.

B.5.4.1 Purpose. The purpose of the Energy Storage Unit Sizing—Flywheels portion of Systems Analysis is to compute the capacity, recurring costs, and non-recurring costs of the flywheels.

B.5.4.2 Background. There has been renewed interest in using flywheels to replace traditional chemical batteries. As flywheel technology matures, the feasibility of using flywheels as mechanical batteries increases. Especially of interest to spacecraft designers is using flywheels in an integrated system to replace both chemical batteries and traditional attitude control systems.

Flywheels function by storing energy in the form of mechanical rotational energy. Energy storage flywheels are required to store large amounts of energy. In order to achieve this with the lowest possible mass, their rotors spin at high RPMs—on the order of tens of thousands to hundreds of thousand of RPMs. These high rotational speeds generate very large axial and tangential forces within the rotors. Therefore, the rotors must be constructed of composite materials. Metallic rotors fail at these high rotation speeds, and they have a greater mass. Some of the new composite rotors have a shaftless design, while others are mounted to shafts. In both instances, magnetic bearings are used to hold the rotor in place. The friction from mechanical (ball) bearings causes excessive energy loss at the desired rotational speeds. However, these new flywheels typically contain backup mechanical bearings in case of magnetic bearing failure. In some designs, mechanical bearings are used during launch when the launch forces exceed the capabilities of magnetic bearings.

Electric motors spin the rotors to store energy while energy is withdrawn using a generator. One piece of hardware typically serves as both the motor and generator. Energy losses in the flywheels stem from drag in the bearings, efficiency losses in the motor/generator, and the steady state energy required to power the bearings and controller.

Storing energy in the form of rotational energy results in flywheels with large angular momentum. Adding or extracting energy from the flywheels changes this angular

momentum. The design of a spacecraft using flywheels for the energy storage system must account for the changes in angular momentum. Additionally, these angular momentum changes can be used for attitude control. This use of the angular momentum for attitude control is described more fully in the attitude control section.

B.5.4.3 Equations. The following equations were used to model the flywheel performance parameters. Equation B.4 calculates the energy storage capacity of the flywheel using the rotor inertia and the minimum and maximum rotational speeds.

$$FlyCapacity = \frac{1}{2 \times 3600} I_s (\omega_{max}^2 - \omega_{min}^2) \quad (B.4)$$

The depth-of-discharge is found using:

$$FlyDOD = 100 \times \left(1 - \frac{\omega_{min}^2}{\omega_{max}^2}\right) \quad (B.5)$$

The flywheel housekeeping power is computed by adding together the magnetic bearing and controller power requirements.

$$FlyPower = ControllerPower + BearingPower \quad (B.6)$$

Using Cost Estimating Relationships (CERs) taken from Space and Missile Systems Center's *Unmanned Space Vehicle Cost Model* (25), the flywheel model computes default non-recurring and recurring costs for each flywheel. The formula uses the Energy Storage Suite, component-level, minimum unbiased percentage error CERs. These CERs provide cost estimates in thousands of fiscal year 1992 dollars (FY92\$K). The model allows for the CER estimates to be adjusted to a different baseline year. The dollar multiplier adjusts costs to the new baseline year.

The non-recurring cost is estimated using the mass of one flywheel. The equation for computing the non-recurring cost for a flywheel not rated for launch is shown below in Equation B.7. The non-recurring cost for a flywheel rated for operation during launch is computed with Equation B.8.

$$FlyNonRecurringCost = (DolMult)(NRCF)(14.523 \times FlyMass \times 2.2) \quad (B.7)$$

$$FlyNonRecurringCost_{LR} = (DolMult)(NRCF) \\ (14.523 \times [(FlyMass + FlyMass_{LR}) \times 2.2]) \quad (B.8)$$

Additionally, the model adjusts non-recurring cost estimates by a development factor taken from Larson (16). This factor is called the non-recurring cost factor (NRCF). Its value depends on the estimated level of research, development, test, and evaluation (RDT&E) required to produce space-qualified flywheel designs. Values greater than one increase the RDT&E estimates, while values less than 1 decrease the estimate. The NRCF estimated for flywheels is 1.5 to account for the lack of experience with, and limited knowledge of, composite flywheel technology.

Recurring, or per unit costs, are computed using the mass of the flywheel. The equation for computing the recurring cost for each flywheel is shown below in Equation B.9 and Equation B.10 for launch-rated flywheels.

$$FlyCost = DolMult \times 7.939(FlyMass \times 2.2)^{0.965} \quad (B.9)$$

$$FlyCost_{LR} = DolMult \times 7.939[(FlyMass + FlyMass_{LR}) \times 2.2]^{0.965} \quad (B.10)$$

B.5.5 Energy Storage Unit Sizing—Chemical Batteries.

B.5.5.1 Purpose. The purpose of Energy Storage Unit Sizing—Chemical Batteries is to determine the mass, volume, reliability, recurring costs, and non-recurring costs of the batteries.

B.5.5.2 Background. Rechargeable chemical batteries are currently the most commonly used method for energy storage on earth-orbiting spacecraft. These chemical batteries are not the primary source of power for the spacecraft; thus they are commonly referred to as “secondary” batteries. Through a reversible chemical reaction, the batteries store energy generated by the solar array during the sunlit portion of the

satellite's orbit. Spacecraft can then rely on batteries for energy during the eclipse portion of the orbit.

In many space applications, the battery chosen for a particular satellite mission is not designed specifically for that mission. Often, engineers choose an off-the-shelf battery for their satellite because that particular battery design has proven performance in the space environment (26). The spacecraft designers then choose the proper number of batteries to meet the energy and lifetime requirements of the mission.

B.5.5.3 Equations. The system-level model uses the method described above to determine the number of batteries required for the mission under evaluation. First, it computes the performance parameters for the particular chemical battery under evaluation based on mission requirements. The SLM then selects the appropriate number of batteries required to meet the mission's energy storage and power requirements. This approach is valid for the two most widely used secondary batteries in space applications: Nickel-Cadmium (Ni-Cd) and Nickel-Hydrogen (Ni-H₂), as well as Li-Ion batteries.

The calculations modeling chemical batteries are described below.

The battery hardware model computes the *Recommended DOD* performance parameter utilizing two equations outlined by Larson (16). These equations compute the recommended depth of discharge based on the type of chemical battery (Ni-Cd, Ni-H₂, or Li-Ion) and the number of charge/discharge cycles (CDCs) predicted for the mission orbit.

For Li-Ion batteries, cycle life does not depend on depth of discharge. Therefore, the *Recommended DOD* is assumed to be 100%. However, note that current Li-Ion technology reports charge/discharge cycle lives of 1500-2000 cycles for Li-Ion batteries. The model is only applicable with Li-Ion batteries for missions requiring less than 2000 charge/discharge cycles (4).

For Nickel-Cadmium batteries, the *Recommended DOD* is found using Equation B.11.

$$\text{Recommended DOD} = -0.314 \log(\text{Total CDC Required}) + 1.562 \quad (\text{B.11})$$

For Nickel-Hydrogen batteries, the *Recommended DOD* is found using Equation B.12.

$$\text{Recommended DOD} = -0.300 \log(\text{Total CDC Required}) + 1.785 \quad (\text{B.12})$$

Total CDC Required is the total number of charge/discharge cycles the satellite is predicted to encounter over the desired mission life.

To compute *Total CDC Required*, the model breaks the orbital altitude range into three regions: low-Earth orbits with altitudes ranging from 200 - 2,500 km, a broad mid-Earth orbit (MEO) region ranging from 2,500 - 35,786 km, and a geosynchronous region at the altitude of 35,786 km. For each of the three altitude regions, the model first computes the predicted number of charge/discharge cycles per year, and then multiplies this number by the *Design Life* to compute *Total CDC Required*.

The model assumes that the spacecraft encounters a charge/discharge cycle each orbit for LEO missions. It computes the number of charge/discharge cycles per year for low-Earth orbits based on the integer number of orbits per day multiplied by 365 days per year.

The model computes the number of CDCs per year for mid-Earth orbits by assuming there are two eclipse seasons per year and each eclipse season averages 50 days in duration (12). It then multiplies the number of orbits per day by 100 eclipse days per year to compute the CDCs per year.

There are 90 charge/discharge cycles per year for geosynchronous orbits. This number is based on one charge/discharge cycle per orbit, one orbit per day, and two 45-day eclipse seasons per year for geosynchronous orbits.

Battery mass and volume are not computed. The values stored in the database are reported along with the other calculated outputs.

The model computes the effective energy storage capacity (*Effective Capacity*) of each battery under consideration. The energy available from a battery with a given rated capacity in amp-hours is the product of the rated capacity multiplied by the 28 V bus voltage. This is a simplifying assumption. The battery voltage will vary throughout the

charge/discharge cycle and is typically not equal to the bus voltage. *Effective Capacity* is calculated using Equation B.13.

$$\begin{aligned} \text{Effective Capacity (Whr)} = & \text{Bus Voltage} \times \text{Rated Capacity} \\ & \times \text{Recommended DOD} \end{aligned} \quad (\text{B.13})$$

The model computes default non-recurring and recurring costs for each battery using CERs taken from Space and Missile Systems Center's *Unmanned Space Vehicle Cost Model* (25). The equation uses the Energy Storage Suite, component-level, minimum unbiased percentage error CERs. These CERs provide non-recurring and recurring cost estimates in FY92\$K. The model allows for the cost estimates to be adjusted to a different baseline year. The dollar multiplier adjusts the costs to the new baseline year.

The non-recurring costs are estimated using the mass of one battery and its rated capacity. The equation for computing the non-recurring costs is shown in Equation B.14.

$$\begin{aligned} \text{Non - Recurring Cost} = & (\text{DolMult} \times \text{NRCF})[115.368 + (2.240 \\ & \times \text{Rated Capacity} \times \text{Battery Mass} \times 2.2)] \end{aligned} \quad (\text{B.14})$$

Similar to the flywheel CERs, the model adjusts non-recurring costs by a development factor which modifies the non-recurring development cost estimates (16). This factor is called the non-recurring cost factor (NRCF). Its value depends on the estimated level of research, development, test, and evaluation (RDT&E) required for a particular battery design. The NRCFs for the three battery types considered are listed below:

1. Ni-Cd = 0.5
2. Ni-H₂ = 0.5
3. Li-Ion = 1.5

The model uses 0.5 for Ni-Cd and Ni-H₂ NRCFs to reflect the off-the-shelf nature of the pre-specified Ni-Cd and Ni-H₂ batteries in the database. These battery technologies are mature, and a full scale RDT&E program would not be necessary. The model uses 1.5 as

the NRCF for the Li-Ion batteries in the database. Li-Ion battery technology is relatively new, and the high NRCF captures this fact by increasing the non-recurring cost estimates above that given by the CER.

Recurring, or per unit, costs are computed using the mass of the battery. The equation for computing the recurring costs for each battery is shown in Equation B.15.

$$\text{Recurring Cost} = (\text{DolMult} \times 7.939)(\text{Battery Mass} \times 2.2)^{0.965} \quad (\text{B.15})$$

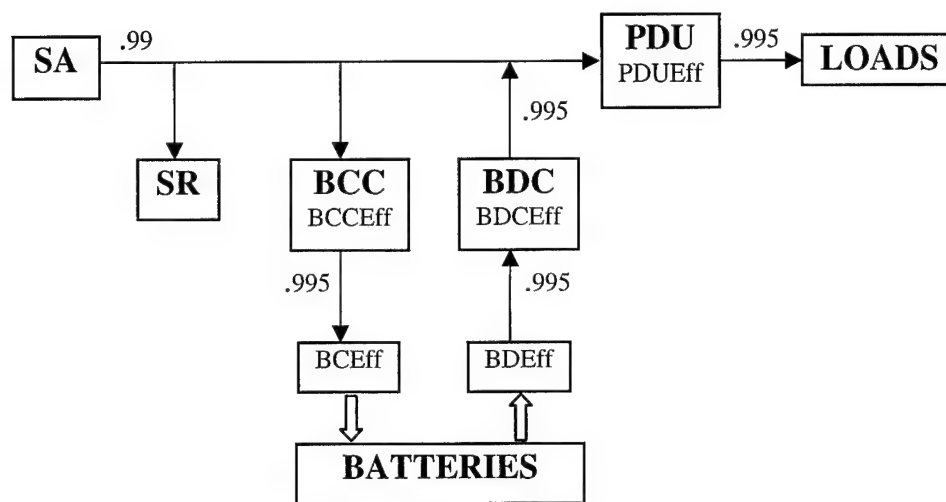
B.5.6 Power Management and Distribution System Model Formulation.

General. The power management and distribution (PMAD) subsystem forms the interface between the energy generation subsystem (EGS), the energy storage subsystem (ESS), and the spacecraft loads. The power management function of this subsystem consists of controlling the storage of both energy and the solar array output, as well as regulating the spacecraft electrical bus voltage. The power distribution function provides for the transmission and switching of electrical power from the spacecraft electrical bus to the individual spacecraft loads. The primary components are the wiring harnesses, power converters, regulators, and power distribution units.

PMAD System Model. There are numerous configurations for PMAD subsystems. To model a PMAD subsystem, it is necessary to choose representative configurations that are typical of most PMAD systems. The model is based on two fundamental architectures. The first uses shunt regulators to regulate the solar array output by dissipating excess energy during sunlight. This approach is termed direct energy transfer (DET). This architecture has the shunt regulator in parallel with the solar arrays and does not introduce inefficiencies due to the regulator's presence. The second architecture uses a peak power tracker (PPT), a type of dc-dc converter, to actively change the operating point of the solar array up to the solar array peak output power. The PPT approach is a non-dissipative means of regulation; the regulator is in series with the solar arrays. The PPT method introduces inefficiencies due to its presence.

Figure B.2 illustrates a DET PMAD system with a Battery ESS. Figure B.3 illustrates a PPT PMAD system with a Battery ESS. Figure B.4 illustrates a DET PMAD system with a Flywheel ESS. Figure B.5 illustrates a PPT PMAD system with a Flywheel ESS.

Battery Energy Storage DET



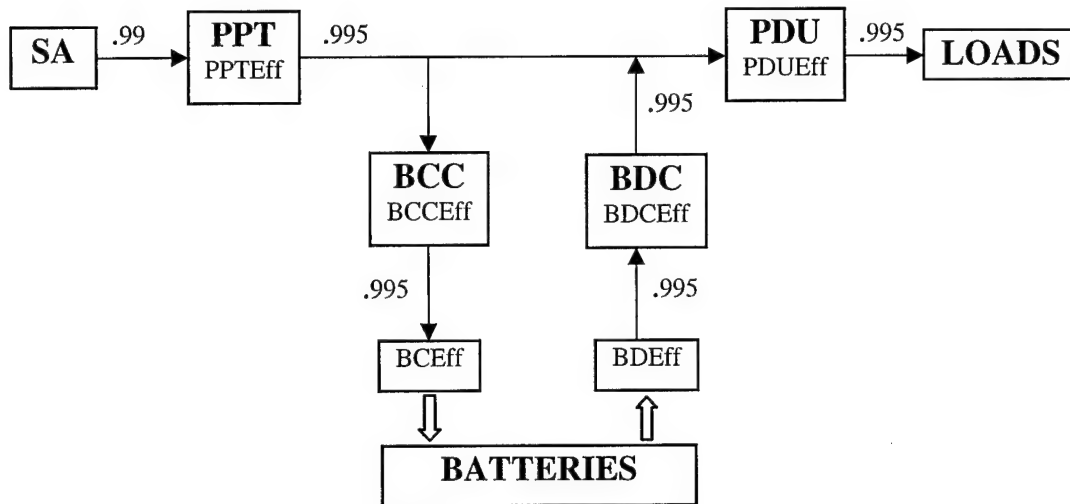
SA - Solar Array
SR - Shunt Regulator
PDU - Power Distribution Unit

BCC - Battery Charge Controller
BDC - Battery Discharge Controller
Eff - Efficiency

Figure B.2 DET PMAD System - Battery ESS

B.5.6.1 Purpose. One objective of the PMAD hardware model is to account for the inefficiencies of the PMAD components. Power between the solar arrays, the ESS, and the loads is passed through the PMAD system. There are losses associated with the flow of power through these components. These losses must be considered when

Battery Energy Storage PPT



SA - Solar Array

PPT - Peak Power Tracker

PDU - Power Distribution Unit

BCC - Battery Charge Controller

BDC - Battery Discharge Controller

Eff - Efficiency

Figure B.3 PPT PMAD System - Battery ESS

sizing the solar arrays and ESS. Another objective of this model is to provide estimates of the relevant PMAD system characteristics. This is done separately for the PMAD architecture with a battery ESS and for the PMAD architecture with a flywheel ESS. The characteristics under consideration are mass, cost, volume, and reliability. The model uses these characteristics as part of the objective measures.

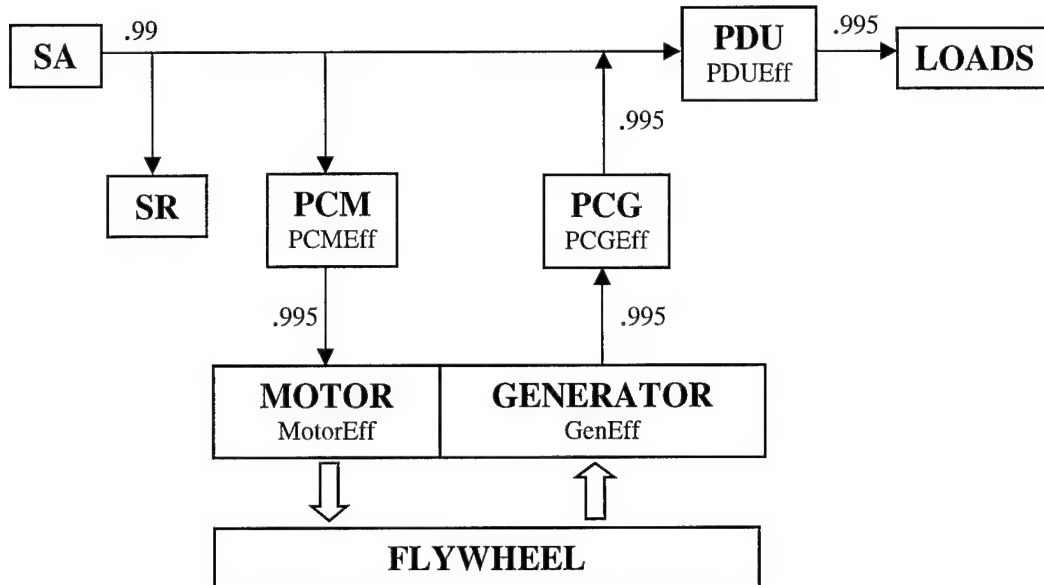
B.5.6.2 Equations. The outputs are produced from the following equations.

Equations - Battery ESS.

- Energy Storage Requirement

$$E_{Req} = \frac{E_{MaxEcl}}{\eta_{Batt-L}} \quad (B.16)$$

Flywheel Energy Storage



SA - Solar Array

SR - Shunt Regulator

PDU - Power Distribution Unit

PCM - Power Converter for Motoring

PCG - Power Converter for Generating

Eff - Efficiency

Figure B.4 DET PMAD System - Flywheel ESS

where

$$\eta_{Batt-L} = 0.995 * \eta_{PDU} * 0.995 * \eta_{BDC} * 0.995 * \eta_{BD} \quad (B.17)$$

The expression η_{Batt-L} represents the total efficiency from the battery output to the load input.

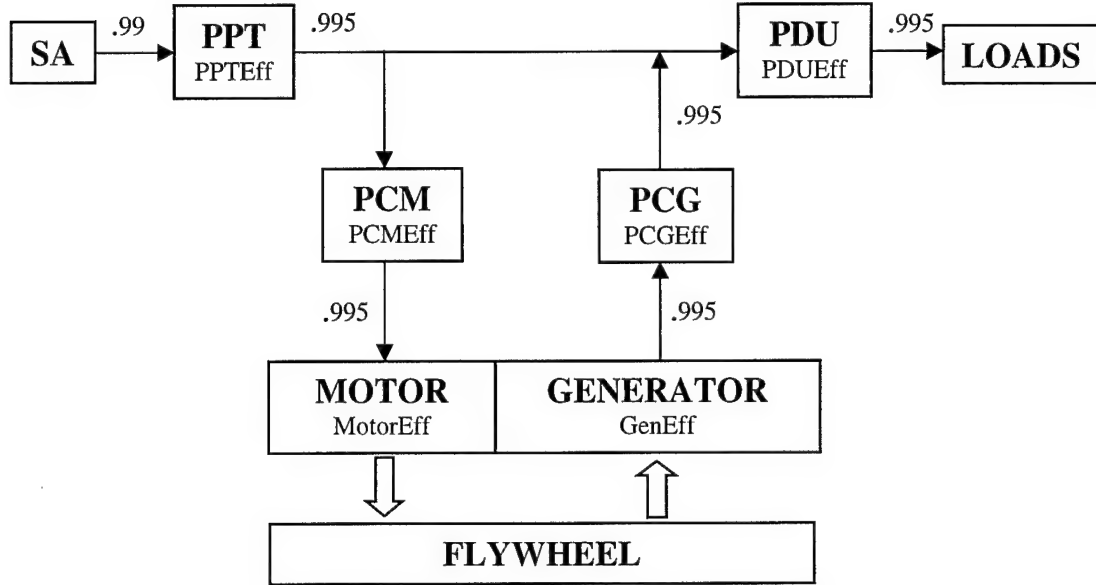
- EOL Power for Energy Storage

$$P_{ES,EOL} = \frac{P_{Batt,EOL}}{\eta_{EG-Batt}} \quad (B.18)$$

where

$$P_{Batt,EOL} = \frac{E_{Req}}{60 * T_{Charge,EOL}} \quad (B.19)$$

Flywheel Energy Storage



SA - Solar Array
PPT - Peak Power Tracker
PDU - Power Distribution Unit
PCM - Power Converter for Motoring
PCG - Power Converter for Generating
Eff - Efficiency

Figure B.5 PPT PMAD System - Flywheel ESS

$$T_{Charge,EOL} = \begin{cases} 15 * T_{MaxEcl} & \text{if } T_{Min} > 15 * T_{MaxEcl} \\ T_{Min} & \text{otherwise} \end{cases} \quad (B.20)$$

$$T_{Min} = T_{Orb} - T_{MaxEcl} \quad (B.21)$$

and

$$\eta_{EG-Batt} = \begin{cases} \eta_{BC} * 0.995 * \eta_{BCC} * 0.99 & \text{if DET} \\ \eta_{BC} * 0.995 * \eta_{BCC} * 0.995 * \eta_{PPT} * 0.99 & \text{if PPT} \end{cases} \quad (B.22)$$

The expression $P_{Batt,EOL}$ represents the power requirement at the battery input for energy storage at EOL. $T_{Charge,EOL}$ is the time available to recharge the batteries at EOL. The expression T_{Min} is the minimum period that the satellite will be in

sunlight. $\eta_{EG-Batt}$ represents the total efficiency from the solar array output to the battery input.

- Satellite Operating Power

$$P_{Op} = \frac{P_1 + P_{ACS}}{\eta_{EG-L}} \quad (B.23)$$

where

$$\eta_{EG-L} = \begin{cases} 0.995 * \eta_{PDU} * 0.99 & \text{if DET} \\ 0.995 * \eta_{PDU} * 0.995 * \eta_{PPT} * 0.99 & \text{if PPT} \end{cases} \quad (B.24)$$

The expression η_{EG-L} represents the total efficiency from the solar array output to the load input.

- PMAD Mass

$$M_{PMAD} = \begin{cases} M_{SR} + M_{BCC} + M_{BDC} + M_{PDU} + M_{Wire} & \text{if DET} \\ M_{PPT} + M_{BCC} + M_{BDC} + M_{PDU} + M_{Wire} & \text{if PPT} \end{cases} \quad (B.25)$$

where

$$M_{SR} = \begin{cases} 0.025 * P_{ES,BOL} * 0.99 & \text{if DET} \\ 0 & \text{if PPT} \end{cases} \quad (B.26)$$

$$M_{PPT} = \begin{cases} 0 & \text{if DET} \\ 0.025 * (P_{ES,BOL} + P_{Op}) * 0.99 & \text{if PPT} \end{cases} \quad (B.27)$$

$$M_{BCC} = \begin{cases} 0.025 * P_{ES,BOL} * 0.99 & \text{if DET} \\ 0.025 * P_{ES,BOL} * 0.99 * \eta_{PPT} * 0.995 & \text{if PPT} \end{cases} \quad (B.28)$$

$$M_{BDC} = 0.025 * \frac{P_1}{0.995 * \eta_{PDU} * 0.995 * \eta_{BDC}} \quad (B.29)$$

$$M_{PDU} = 0.02 * \frac{P_1}{0.995 * \eta_{PDU}} \quad (B.30)$$

$$M_{Wire} = 0.025 * M_{Dry} \quad (B.31)$$

The expressions M_{SR} , M_{PPT} , M_{BCC} , M_{BDC} , M_{PDU} , and M_{Wire} represent the mass of the Shunt Regulator (SR), PPT, BCC, BDC, PDU, and wiring harnesses, respectively.

- PMAD Cost

$$C_{PMAD,NR} = 416.033 + 23.754 * (2.203 * M_{PMAD}) \quad (B.32)$$

and

$$C_{PMAD,R} = 55.984 * (2.203 * M_{PMAD})^{0.698} \quad (B.33)$$

The expressions $C_{PMAD,NR}$ and $C_{PMAD,R}$ represent the non-recurring costs and the recurring costs of each unit, respectively.

- PMAD Volume

The volume estimates, based on the characteristics of actual hardware from Mostert (22), provide a best approximation of system volume.

$$V_{PMAD} = 0.01 + 0.01 + 2 * 0.0008 * N_{Batt} \quad (B.34)$$

The first two terms represent the volume of the shunt regulator (DET) or PPT, and the PDU. The last term represents the volume of the BCCs and BDCs given N_{Batt} .

Equations - Flywheel ESS.

- Energy Storage Requirement

$$E_{Req} = \frac{E_{MaxEcl}}{\eta_{Fly-L}} \quad (B.35)$$

where

$$\eta_{Fly-L} = 0.995 * \eta_{PDU} * 0.995 * \eta_{PCG} * 0.995 * \eta_{Gen} \quad (B.36)$$

The expression η_{Fly-L} represents the total efficiency from the flywheel output to the load input.

- Energy Generation Output Power Requirement at EOL

$$P_{EG,EOL} = \frac{P_{Fly}}{\eta_{EG-Fly-L}} \quad (B.37)$$

where

$$P_{Fly} = \frac{\sum_{i=1}^5 P_i * T_i + E_{GEO} * 60 + E_{Slew} * 60 + P_{Wheel} * T_{Orb}}{T_{Min}} \quad (B.38)$$

and

$$\eta_{EG-Fly-L} = \begin{cases} 0.995 * \eta_{PDU} * 0.995 * \eta_{PCG} * 0.995 * \eta_{Gen} * \\ \quad \eta_{Mot} * 0.995 * \eta_{PCM} * 0.99 & \text{if DET} \\ 0.995 * \eta_{PDU} * 0.995 * \eta_{PCG} * 0.995 * \eta_{Gen} * \\ \quad \eta_{Mot} * 0.995 * \eta_{PCM} * 0.995 * \eta_{PPT} * 0.99 & \text{if PPT} \end{cases} \quad (B.39)$$

P_{Fly} is the load and flywheel input power required to provide energy to the loads and store E_{MaxEcl} during T_{Min} . The expression $\eta_{EG-Fly-L}$ represents the total efficiency from the solar array output through the flywheels to the load input.

- Total Efficiency for Power Transfer Between Wheels

$$\eta_{Fly} = \eta_{Gen} * 0.995 * \eta_{PCG} * 0.995 * \eta_{PCM} * 0.995 * \eta_{Mot} \quad (B.40)$$

- Total Efficiency from Energy Generation Output to Flywheel Input

$$\eta_{EG-ES} = \begin{cases} 0.99 * \eta_{PCM} * 0.995 * \eta_{Mot} & \text{if DET} \\ 0.99 * \eta_{PPT} * 0.995 * \eta_{PCM} * 0.995 * \eta_{Mot} & \text{if PPT} \end{cases} \quad (B.41)$$

- Total Efficiency from Energy Generation Output to Load Input

$$\eta_{EG-L} = \begin{cases} 0.99 * \eta_{PDU} * 0.995 & \text{if DET} \\ 0.99 * \eta_{PPT} * 0.995 * \eta_{PDU} * 0.995 & \text{if PPT} \end{cases} \quad (B.42)$$

- PMAD Mass

Conservative mass estimates based on the power being converted or switched by the PMAD components are used in determining system mass. The equations are from Larson (18) and provide a best approximation.

$$M_{PMAD} = \begin{cases} M_{SR} + M_{PCM} + M_{PCG} + M_{PDU} + M_{Wire} & \text{if DET} \\ M_{PPT} + M_{PCM} + M_{PCG} + M_{PDU} + M_{Wire} & \text{if PPT} \end{cases} \quad (B.43)$$

where

$$M_{SR} = \begin{cases} 0.025 * P_{ES,Fly} * 0.99 & \text{if DET} \\ 0 & \text{if PPT} \end{cases} \quad (B.44)$$

$$M_{PPT} = \begin{cases} 0 & \text{if DET} \\ 0.025 * P_{EG,BOL} * 0.99 & \text{if PPT} \end{cases} \quad (B.45)$$

$$M_{PCM} = \begin{cases} 0.025 * P_{ES,Fly} * 0.99 & \text{if DET} \\ 0.025 * P_{ES,Fly} * 0.99 * \eta_{PPT} * 0.995 & \text{if PPT} \end{cases} \quad (B.46)$$

$$M_{PCG} = 0.025 * \frac{P_1}{0.995 * \eta_{PDU} * 0.995 * \eta_{PCG}} \quad (B.47)$$

where

$$P_{ES,Fly} = P_{EG,BOL} - \frac{P_{Min}}{\eta_{EG-L}} \quad (B.48)$$

$P_{ES,Fly}$ is the maximum power from the solar array output that can be directed to the flywheels. This energy can be stored in the flywheels, or it can be shunted. The

equations for M_{PDU} and M_{Wire} are the same as for a PMAD system with a battery ESS. The expressions M_{PCM} and M_{PCG} represent the mass estimates for the PCM and the PCG, respectively.

- PMAD Cost

$$C_{PMAD,NR} = 416.033 + 23.754 * (2.203 * M_{PMAD}) \quad (B.49)$$

and

$$C_{PMAD,R} = 55.984 * (2.203 * M_{PMAD})^{0.698} \quad (B.50)$$

The expressions $C_{PMAD,NR}$ and $C_{PMAD,R}$ have the same interpretation given FES as they do for a PMAD system with a battery ESS.

The cost estimates, from the Weight-Based Cost Estimating Relationships (CER) in Nguyen (24), are used in determining system costs and provide a best approximation to PMAD system non-recurring and recurring costs.

- PMAD Volume

$$V_{PMAD} = 0.01 + 0.01 + 2 * 0.0008 * N_{Fly} \quad (B.51)$$

The first two terms represent the volume of the shunt regulator (DET) or PPT, and the PDU. The last term represents the volume of the PCMs and PCGs given N_{Fly} .

B.5.7 Energy Generation.

B.5.7.1 Purpose. The purpose of the energy generation model is to determine the mass, volume, reliability, recurring costs, and non-recurring costs of the energy generation subsystem.

B.5.7.2 Equations. Other equations in the model compute power requirements the energy generation subsystem (EGS) must satisfy. This power level, $PwrThru$, must be generated by the solar array while in sunlight at end-of-life (EOL).

Given a battery energy storage system, this power requirement, P_{wrThru} , is the sum of P_{OpBatt} , the operating power PMAD needs from EG for loads at EOL, and $P_{RpEOLBatt}$, the recharge power PMAD needs from EG to recharge batteries at EOL.

Given a FES system, the power requirement, P_{wrThru} , is $P_{TotEOLFly}$, the total power PMAD needs from EG at EOL.

B.5.7.3 Compute Solar Array Area. The following equations are used to determine the area of the solar array.

The mounting coefficient, $MountingCoef$, is the ratio of the actual area of the solar array to its effective area, the area effectively receiving direct sunlight.

- For a panel-mounted, planar array, $MountingCoef = 1$.
- For a body-mounted, cylindrical array, $MountingCoef = 2.5$. This is lower than would be expected given the geometry of cylindrical arrays. Performance increases when the solar array cools as it spins into and out of sunlight (16).
- For a body-mounted, omni-directional (spherical) array, $MountingCoef = 4$.

The temperature of a solar array is a function of altitude. It typically ranges from 67 deg C in LEO to 53 deg C in GEO (16). Assuming a linear interpolation,

$$ArrayTemp (deg C) = -0.0004 \times Altitude + 67.3 + 5(\text{if body-mounted}) \quad (B.52)$$

The temperature inefficiency of the solar array material is η_{TempSA} . This is the reduction in array performance experienced for every degree Celsius the temperature of the array exceeds 28 deg C (16). The temperature inefficiency of the array is

$$TempIEff = \eta_{TempSA} \times (ArrayTemp - 28) \quad (B.53)$$

The efficiency of the solar array material is $ArrayEff$. Traditional values range from 11% to 18% for silicon and gallium arsenide, respectively. The solar array must be sized to meet power requirements when solar radiation intensity is at a minimum. This

occurs at aphelion on July 4th, when Earth is farthest from the Sun. The minimal solar intensity experienced, $MinSInt$, is 1,309 W/m² (2). Therefore, the minimal areal power density of the array, the smallest power produced per unit area if sunlight strikes all of the solar cells, is

$$ArealPwrDen (W/m^2) = (1309 W/m^2) \times ArrayEff \quad (B.54)$$

The efficiency of the solar array related to design, assembly, shadowing, and optical losses of the array is η_{DesSA} . The overall inefficiency of the solar array is

$$TotalInEff = 1 - [\eta_{DesSA}(1 - TempIEff)] \quad (B.55)$$

The maximum angle of incidence of sunlight on the solar array is $MaxAngInc$. The areal power density of the array at BOL, with efficiencies and incidence incorporated, is

$$\begin{aligned} ArealPwrDenBOL (W/m^2) &= ArealPwrDen \times (1 - TotalInEff) \\ &\times \cos(MaxAngInc) \end{aligned} \quad (B.56)$$

The mission design life of the satellite is *Design Life*, and the annual degradation of the array due to meteors and solar radiation is $AnnDegs_A$. The lifetime degradation of the array becomes

$$LifeDegrad = (1 - AnnDegs_A)^{Design Life} \quad (B.57)$$

The areal power density at end-of-life becomes

$$ArealPwrDenEOL (W/m^2) = ArealPwrDenBOL \times LifeDegrad \quad (B.58)$$

The area of the solar array, $ArrayArea$, can then be computed using the power PMAD requires from the energy generation subsystem, $PwrThru$.

$$ArrayArea (m^2) = MountingCoef \frac{PwrThru}{ArealPwrDenEOL} \quad (B.59)$$

B.5.7.4 Solar Array Volume. According to Agrawal (2), typical diameters of solar array booms range from 2.2 and 3.4 cm. Averaging these values and adding 1.3 cm of thickness for the array itself and the coverglass, we estimate the total thickness of the array to be approximately 4 cm. This value also works for body-mounted arrays, as the mounting would use space not used for the booms. The volume of the array can then be computed as

$$ArrayVol (m^3) = 0.04 \times ArrayArea \quad (B.60)$$

B.5.7.5 Compute Solar Array Mass. The following equations are used to determine the mass of the solar array.

The specific power of the solar array material is *ArraySP*. The specific power of the array at BOL can be computed as follows:

$$\begin{aligned} SASpPwrBOL (W/kg) = & ArraySP \times (1 - TotalInEff) \\ & \times \cos(MaxAngInc) \end{aligned} \quad (B.61)$$

Specific power at EOL, *SASpPwrEOL*, takes into account the degradation over the array's lifetime.

$$SASpPwrEOL (W/kg) = SASpPwrBOL \times LifeDegrad \quad (B.62)$$

The solar array's mass is then

$$ArrayMass (kg) = \frac{PwrThru}{SASpPwrEOL} \quad (B.63)$$

B.5.7.6 Solar Array Reliability. Our research did not uncover any information regarding the probability a solar array will still be performing its function at the satellite's end-of-life. It appears to be generally assumed that a catastrophic failure is very unlikely. For these reasons, the energy generation model reports a solar array reliability, *ArrayRel* = 0.999.

B.5.7.7 Solar Array Recurring and Non-recurring Costs. CERs for solar arrays were determined by Air Force Materiel Command's Space and Missile Systems Center (25). These CERs estimate the recurring and non-recurring costs for a solar array as a function of the array's mass, its power output at beginning-of-life, and the total number of solar cells. The size of the solar cells is assumed to be 4 cm x 6 cm.

The number of solar cells, $NumSCells$, is

$$NumSCells = \frac{ArrayArea}{0.0024 \text{ m}^2} \quad (B.64)$$

According to Space and Missile Systems Center, the recurring cost is

$$ArrayCost (FY92\$K) = 4.959 \times NumSCells^{0.621} \quad (B.65)$$

The non-recurring cost is

$$ArrayNRCost (FY92\$K) = 0.024 \times ArrayMass \times \frac{PwrThru}{LifeDegrad} \quad (B.66)$$

$$+ 0.021 \times NumSCells \quad (B.67)$$

B.5.8 Attitude Control Module Formulation.

B.5.8.1 Purpose. Modeling the attitude control function consists of developing seven measures to predict relative performance of an attitude control system (ACS). These measures are torque capability, momentum storage, mass, non-recurring cost, recurring cost, volume, and reliability. We looked at three different methods of performing attitude control for a satellite with flywheel energy storage (FES). These methods were momentum wheel approach (MWA), gimbale flywheel approach (GFA), and a separate, traditional attitude control approach (TACA). The first two methods are considered integrated power and attitude control systems (IPACS). The third method uses flywheels for energy storage only, while the TACA reacts to changes in the total flywheel angular momentum and handles external torques.

B.5.8.2 Equations. The model provides results for the seven performance measures and calculates power and energy requirements for other sections of the model.

1. Power Requirement

Power budgets are an important part of spacecraft design. The spacecraft is limited to the energy stored by the ESS and the energy provided by the solar arrays. In general, it is important to minimize the power requirements of subsystems. The three ACS methods have different power requirements and different ways of measuring those power requirements. Energy requirements are used in determining the impact of the different ACS methods on the number of energy storage units required.

MWA This method stores energy without affecting attitude, assuming perfect control of the flywheel is possible. In real systems, energy storage would cause small torques due to imperfect control. Power requirements are the efficiency losses as energy is moved from one flywheel to another to maintain attitude. During energy storage and energy removal, these attitude control losses may not occur if all the torques to store or remove energy are greater than the required attitude torques. There would still be efficiency losses due to storing or removing energy, but we account for these in a separate part of the SLM.

Hall shows that the minimum number of wheels necessary to store energy and control attitude using the MWA is 4 wheels (10). With N wheels, where N is the number of wheels, there are 3 torque modes in attitude control which also affect energy storage, and $(N - 3)$ torque modes in energy storage which do not affect attitude control. These energy storage modes are the null space of the flywheel spin axes. The torques for controlling attitude changes the amount of energy stored in the flywheels. Given an amount of energy to be stored in or removed from the flywheels, the remainder is stored or removed using the energy storage modes. The following three examples demonstrate these ideas.

(a) Null space example

A spacecraft has 4 flywheels. One is aligned with the spacecraft's x-axis. The second is aligned with the y-axis. The third is aligned along $-0.6\hat{i} + 0.8\hat{k}$. The last wheel is aligned along $-0.8\hat{j} - 0.6\hat{k}$. The matrix of spin axes is as follows:

$$A = \begin{bmatrix} 1 & 0 & -0.6 & 0 \\ 0 & 1 & 0 & -0.8 \\ 0 & 0 & 0.8 & -0.6 \end{bmatrix}$$

The null space of the attitude control is the ratio of change in wheel speeds that cause no change in angular momentum. For this example, the null space is

$$\Delta w = \begin{bmatrix} 0.45 \\ 0.8 \\ 0.75 \\ 1 \end{bmatrix}$$

By changing wheel speeds in multiples of this vector, the angular momentum of the spacecraft will not change. Since there are four wheels, there is only one energy mode. Therefore, any other combination of changes in wheel speeds will cause a change in the angular momentum and attitude of the satellite. A flywheel energy storage system can take advantage of this fact. Hall (10) shows how to decompose the wheel speeds, angular momentum and the torques down into energy and attitude control components using singular value decomposition. The torques on the wheel decompose into a linear combination of the attitude control torque and the energy storage torque. As long as four non-coplanar wheels are operational, the flywheels can store energy and maintain attitude.

(b) Zero power requirement

Assume 4 wheels in a tetrahedral configuration. The solar arrays are providing 4 kW to the flywheels. Each flywheel is rotating at 2500 rad/s and

is being charged with 1 kW of power. The necessary torque to provide that power is 0.4 Nm to each flywheel. If the motor efficiency, $\eta_{motor} = 0.95$, and no attitude control torques are required, the losses are 200 W. If the attitude control torque required is -0.1 Nm to a single wheel, then the torques necessary to store 4 kW are [0.325 0.425 0.425 0.425] Nm. The associated total losses are

$$(1 - 0.95) * 2500\text{rad/s} * (0.325 + 0.425 + 0.425 + 0.425)\text{Nm} = 200\text{W}$$

The ACS in this example requires no additional power to operate.

(c) ACS operating power

Assume 4 wheels in a tetrahedral configuration. All are currently spinning at 5000 rad/s. The solar arrays are providing sufficient power to the loads and are not currently charging the flywheels. The null space of the attitude control, the energy storage mode, is changing all the wheel speeds equally. Torquing a single wheel with +0.1 Nm can satisfy the attitude control torque. The attitude control torque provides 500W. Since the flywheels are not providing power to a load, nor receiving power, the power would be stored in the flywheel energy storage. To keep the energy in the system constant, each wheel, including the wheel satisfying the attitude control torque, must be torqued by -0.025 Nm. The net change to a 100% efficient system is 0 W of power, and [0.075 -0.025 -0.025 -0.025] Nm of torque. In a real system, efficiency losses are due to the interwheel efficiencies, assumed in this case to be $\eta_{iw} = 0.90$. The losses are the power requirement of the ACS,

$$(1 - \eta_{iw})\left(\frac{N-1}{N}T_{ext} - T_{en}\right)\omega = (1 - 0.9)\left(\frac{3}{4}0.1 - 0\right)\text{Nm}(5000\text{rad/s}) = 37.5\text{W}$$

The result is that if the torques associated with energy storage or removal are greater than the attitude control torques, there are no additional losses due to controlling attitude. However, throughout an orbit, there may be

cases where the attitude control torques are greater than the energy storage torques, especially when the flywheels are near full charge. This will result in energy being transferred from one wheel to another, rather than coming from the solar array or going to the loads. The energy requirement for the attitude control is then the total energy transferred between wheels over an orbit multiplied by the interwheel inefficiency.

Calculating the worst-case energy requirement for the MWA is complicated. The process begins with the solar array and the orbit. At LEO, the eclipse period is a relatively large portion of the orbit. The spacecraft uses flywheels to provide power during eclipse. During the sunlit period, the solar array provides enough power to cover loads and recharge the flywheels. The attitude control system requires the most power if the torque's direction opposes, and its magnitude exceeds, the energy storage torques. To calculate the worst-case power requirements, the external torques are always assumed to oppose the energy torques. The greatest energy use for attitude control during eclipse occurs when loads are a minimum, and therefore energy torques are a minimum. A low energy torque maximizes the likelihood that energy is passed between wheels. This transfer is needed to maintain attitude. The low energy torque also means more energy is passed. However, the solar array in our model is designed to provide for maximum loads during eclipse. Since the power requirement for ACS is an efficiency loss, the change in power from minimum to maximum load is greater than the decrease in ACS power. Thus, the maximum total eclipse energy coincides with maximum loads during eclipse. This is the case we consider in our model.

To calculate the power requirement, we assume that adding energy evenly to all of the flywheels spinning at the same speed will not affect attitude control. In other words, the energy storage mode is a vector of all ones. Energy provided to the load during load period i by each flywheel is P_i/N , and the power leaving each flywheel is

$$P_{Fi} = \frac{P_i}{N\eta_{ES-L}} \quad (B.68)$$

The total rotational energy in a flywheel is

$$E = I_s \omega^2 \quad (B.69)$$

At the end of the load period or eclipse, the new wheel speed is

$$\omega_{end} = \sqrt{\omega_{beg}^2 - \frac{P_F LD}{I_s}} \quad (B.70)$$

The torque on the flywheel due to storing or removing energy, if the external torques are zero, is

$$T_{en} = \frac{P_F}{\omega} \quad (B.71)$$

The torque during eclipse increases as the wheel is spun down to provide constant power. Therefore, a comparison of the energy torque to the external torque is made at the beginning and at the end of a load period. This is a three dimensional problem. When the external torque barely exceeds the energy torque in a wheel, the worst-case power requirement occurs if the external torque aligns with a single wheel. When no energy torques were present, the maximum power requirement occurs when three orthogonal flywheels must provide equal torque to counter the external torque. There are other options for systems with more flywheels and more complex arrangements. The simplest case is when the external torque aligns with a single flywheel axis. As mentioned above, this is the worst-case ACS power in certain cases. This will be explained in more detail shortly. For the other cases, we define a power multiplier, X, which is used to calculate the worst-case ACS power. X depends on the external torque and the torque to store or remove energy from the flywheels.

The development of this power multiplier begins by assuming orthogonal wheels are used to control the attitude. This disagrees with the assumption of a zero

momentum arrangement for less than 6 wheels. However, orientation changes with every added wheel. Other results would follow from different orientations, but for the purposes of comparing different wheels, this method sufficed. This method gave a quick approximation of the worst-case power requirement without trying to optimize the configuration for every alternative. There are three cases to examine where power flows between wheels:

Single axis – The external torque is aligned with a single flywheel's spin axis.

The external torque's energy can be canceled by torquing every wheel equally in the opposite direction with a torque equal to $\frac{T_{ext}}{N}$. Because every wheel must be torqued to cancel the external torque's energy, each flywheel countering the external torque is torquing $\frac{N-1}{N}T_{ext}$ before storing energy. Therefore $\frac{N-1}{N}T_{ext}$ must exceed T_{en} to cause energy to switch between wheels. The power flowing between wheels, P_{bw} , is:

$$P_{bw_1} = \max\left(\frac{N-1}{N}T_{ext} - T_{en}, 0\right)\omega \quad (B.72)$$

Two axes – The external torque is coplanar with two attitude spin axes. In this case, the maximum power loss is when T_{ext} is split evenly between the two axes.

$$\hat{T}_{ext} = \frac{\sqrt{2}}{2}T_{ext}(\hat{i}_1 + \hat{i}_2) \quad (B.73)$$

A second option is the degenerative case of all the torque in a single axis. In this two-axis case, every wheel must be torqued $\frac{\sqrt{2}}{N}T_{ext}$. T_{ext} must exceed $\frac{N}{N-2}\sqrt{2}T_{en}$ to cause energy to switch between wheels. However, when

$$T_{ext} > \frac{N}{(\sqrt{2}-1)N + 1 - 2\sqrt{2}}T_{en} \quad (B.74)$$

two axis alignment transfers more power between wheels than one axis alignment. This will only be true for 5 or more flywheels. The power flowing between wheels in the two axis case is:

$$P_{bw_2} = 2 \max\left(\frac{(N-2)\sqrt{2}}{2N}T_{ext} - T_{en}, 0\right)\omega \quad (B.75)$$

Three axes – The external torque is not aligned with any attitude spin axes.

Through the same arguments, the maximum power loss due to an external torque in three attitude control axes is when the torque is split evenly between all three axes. T_{ext} must exceed $\frac{3N}{N-3}\sqrt{3} T_{en}$ to cause energy to switch between wheels. When

$$T_{ext} > \frac{N}{(\sqrt{3} - \sqrt{2})N + 2\sqrt{2} - 2\sqrt{3}}T_{en} \quad (B.76)$$

three axis alignment transfers more power between wheels than two axis alignment. This will only be true only for 8 or more flywheels. With this many flywheels, the flywheels can be oriented to have at least three energy storage modes that change some or all of the flywheels speeds equally. With this adjustment, only the first two power multipliers need to be considered. The power transfer in this final case is:

$$P_{bw_3} = 3 \max\left(\frac{(N-3)\sqrt{3}}{3N}T_{ext} - T_{en}, 0\right)\omega \quad (B.77)$$

The power requirement for attitude control is the maximum efficiency loss for moving energy from wheel to another.

$$P_{MWA} = (1 - \eta_{IW})\left(\frac{N - X^2}{N}X T_{ext} - X^2 T_{en}\right)\omega \quad (B.78)$$

where X depends on the set of axes that cause the greatest energy transfer. Using X allows a single equation to be developed.

$$X = \begin{cases} 0 & \text{if } \frac{N}{N-1}T_{en} > T_{ext} \\ 1 & \frac{N}{N-1}T_{en} < T_{ext} < \frac{N}{(\sqrt{2}-1)N+1-2\sqrt{2}}T_{en} \\ \sqrt{2} & \frac{N}{(\sqrt{2}-1)N+1-2\sqrt{2}}T_{en} < T_{ext} \end{cases} \quad (B.79)$$

If the beginning and ending torques fall into different power multiplier categories, the changeover point is easily calculated from the following formulas:

$$F_{ch} = \begin{cases} \frac{N-1}{N} & \text{if } X_{beg} = 1 \\ \frac{(\sqrt{2}-1)N+1-2\sqrt{2}}{N} & X_{beg} = \sqrt{2} \end{cases} \quad (B.80)$$

$$\omega_{ch} = \frac{P_F(N-1)}{F_{ch}NT_{ext}} \quad (B.81)$$

$$t_{ch} = \frac{\omega_{beg}^2 - \omega_{ch}^2}{\omega_{beg}^2 - \omega_{end}^2} LD \quad (B.82)$$

The final set of equations calculates the maximum eclipse energy. The attitude energy requirement during a load period comes in two parts if X changes over the load period. The result was zero unless X is nonzero. When X is nonzero, the external torques are large enough to require energy transfers from one wheel to another during eclipse.

The development begins by assuming a constant load on the flywheel, P_F . The attitude control energy requirement is the integral of the power from Equation B.78 from $t = t_{beg}$ to $t = t_{end}$, or $t = t_{ch}$, if it exists. The values of X and T_{ext} remain constant over the integral. Therefore, the integral of equation B.78 can be easily separated into two separate integrals. $T_{en}\omega$ is always P_F . The first integral requires more work. We know that

$$\omega(t) = \sqrt{\omega_{beg}^2 - \frac{P_F t}{I_s}} \quad (B.83)$$

Integrating results in

$$\int_0^t \frac{N - X^2}{N} X T_{ext} (1 - \eta_{IW}) \omega dt = \frac{2I_s(\omega_{beg}^3 - \omega_{end}^3)(1 - \eta_{IW}) \frac{N-X^2}{N} X T_{ext}}{3P_F} \quad (B.84)$$

Putting the two integrals together produce the attitude control energy requirement for a fixed X and P_F .

$$E_{ACS} = \sum_{load\ pds} \sum_X (1 - \eta_{IW}) \left(\frac{2(N - X^2)X T_{ext} I_s (\omega_{beg}^3 - \omega_{end}^3)}{3NP_F} - X^2 P_F LD \right) \quad (B.85)$$

The next energy requirement is due to any slew maneuvers. The maximum energy for a slew maneuver is

$$E_{eclslew} = \sqrt{3} g_{slew} \omega_{max} (1 - \eta_{IW}) \min(t_{slew} N_{slew}, E_{clmax}) \quad (B.86)$$

The solar array must provide this additional ACS energy during the recharge period.

At GEO, the method for storing energy follows a different path. Rather than an eclipse every orbit, eclipse seasons include orbits during which no eclipse occurs. Also, eclipse durations never exceed 6% of the orbital period. The solar array provides sufficient power to cover the spacecraft's energy needs over the entire orbit. The flywheels provide the excess power needed during peak requirements in addition to powering the spacecraft during eclipses. This results in lower torques on the wheels due to energy storage, which increases the attitude control power usage. The largest power requirement occurs when the wheels are operating near maximum speed. We used the following formula to estimate the smallest torque necessary to perform energy storage.

$$T_{en} = \begin{cases} \frac{\eta_{EG-ES}(\eta_{EG-L} P_{SA} - P)}{\omega_{max} N} & \text{if } P < \eta_{EG-L} P_{SA} \\ \frac{\eta_{ES-L}(P - \eta_{EG-L} P_{SA})}{\omega_{max} N} & \text{otherwise} \end{cases} \quad (B.87)$$

Summing over the load periods gives the worst-case energy requirement for attitude control over an orbit. (See Equation B.78.) Slew maneuvers account for additional energy according to Equation B.86.

GFA This method requires an additional load to power the gimbals, which must be moved to counter energy storage or removal in the wheels as well as external

torques. However, GFA does not require energy transfer from one flywheel to another.

The gimbal power requirements are from Larson (16), where the steady state power per gimbal is 15 to 30 W in stand-by mode and an additional 0.02 to 0.2 W/Nm when torquing. The following formula provides a first estimate for modeling gimbaleed flywheels using the magnetic bearings to provide the torque.

$$P_{gimbal} = 20W \text{ standby} + 0.1W/Nm \text{ when torquing} \quad (B.88)$$

TACA Power requirements are similar to the GFA listed above. The flywheels used only for energy storage are counter-rotated to cancel momentum and torque effects of storing and removing energy from the flywheels. Power requirements are derived from the off-the-shelf hardware information. For the ACS model, actual attitude control system data is used.

2. Torque Capability

This output is a function of the hardware, external torques, stability requirements, slew rates, and flywheel energy storage torques. More torque capability for attitude control provides better stability and better slew rates. The minimum torque requirement is the higher of

- the slew torque requirement, or
- the sum of: external torques, the torque necessary to maintain stability, and the torques due to flywheel energy storage

The minimum torque requirement is assumed to be possible in any axis. The output torque capability is the torque capability of the ACS minus the minimum torque requirement.

All of the following torque equations are from Larson (16).

The equation for the stability torque is derived for separation from the booster and assumes a tip-off rate which is the angular velocity at separation. The ACS requires a minimum torque to keep the spacecraft within the maximum attitude deviation. A

maximum body spin rate is assumed, using a linear formula with respect to altitude. The outcome is a reasonable approximation of the necessary stability torques. Larger results, over 0.2 Nm, are much less accurate and should be examined more closely. The slew torque equation assumes a full acceleration torque for half the maneuver and a full deceleration torque for the remaining time. The external torques that must be overcome include gravity gradient, aerodynamic, magnetic, and solar pressure. The resulting external torque is the sum of the four external torques.

$$T_{stab} = \frac{\pi \omega_{maxbodyspinrate}^2 max(I_x, I_y, I_z)}{360 \theta_{maxdev}} \quad (B.89)$$

$$\omega_{maxbodyspinrate} = 0.0015 + (42164 - r)/4000000 rad/s \quad (B.90)$$

$$T_{sl} = \frac{4 max(I_x, I_y, I_z) \theta_{slew}}{t_{slew}^2} \quad (B.91)$$

$$T_{gg} = \frac{3\mu(I_z - min(I_x, I_y)) \sin 2\theta_{maxdev}}{2r^3} \quad (B.92)$$

$$T_{ggslew} = \frac{3\mu(I_z - min(I_x, I_y)) \sin 2\theta_{slew}}{2r^3} \quad (B.93)$$

$$T_{aero} = 0.5 \rho C_d A V^2 (c_{pa} - c_g) \quad (B.94)$$

$$T_{mag} = \frac{F_I D M}{r^3} \quad (B.95)$$

$$T_{solar} = \frac{K A (1 + q) (c_{ps} - c_g)}{c} \quad (B.96)$$

$$T_{ext} = T_{ggslew} + T_{aero} + T_{mag} + T_{solar} \quad (B.97)$$

MWA With the momentum wheel approach, the ACS provide torques by changing flywheel speeds. Flywheels operate in roughly the same range as reaction wheels. It is the motor/generator that determines the limit of the torque possible. Torques for reaction wheels are typically from 0.03 to 2 Nm. These values are indicative of current reaction wheels and depend on the motor/generator. The MWA is assumed to be a zero momentum system, where all of the wheels are spinning at the same speed. Thus the total angular momentum of the flywheels is zero. This method requires a minimum of 4 wheels to provide both energy storage and attitude control. Four wheels are assumed to provide the torque capability of a single flywheel in each axis. To estimate the effect of

additional wheels beyond the minimum of 4, extra wheels in groups of three provide the additional torque capability of a single wheel in each axis. Using less than an additional three wheels is estimated by assuming the wheel provides one-third of its torque capability in each primary axis. Since adding a wheel requires a change of orientation, this assumption is close but not exact.

$$T_{cap} = \frac{N-1}{3}T_{mg} - \max[T_{stab} + T_{ext} + Res \max(T_{en}), T_{st}] \quad (B.98)$$

GFA With the gimbaled flywheel approach, the ACS provides torques by changing the spin axes of the flywheel with respect to the spacecraft using a gimbal. Torques of control moment gyros (CMGs) are typically 25 to 500 Nm. For flywheels, the angular momentum is higher, due to the high RPMs of flywheels in comparison to current CMGs. The increase in angular momentum allows a traditionally gimbaled flywheel to provide the same torque for a smaller gimbal deflection. Flywheels also have the option of magnetically gimbaling the flywheel. This method saves on mass, but makes the magnetic bearings more complicated. The equation approximates the gimbal torque output based on an assumed gimbal rate. The torque created by storing energy in the flywheels does not cancel in this method. The GFA needs to counter the full energy storage torque. This method requires a minimum of 2 wheels to provide both energy storage and attitude control. However, using only 2 wheels requires a mixture of changing wheel speeds and spin axes. Therefore, for the GFA, the assumption is that wheel speeds are used strictly for energy storage, and changing the spin axes is the only method controlling attitude. Three wheels are assumed to provide the torque capability of a single flywheel in each axis. To estimate the effect of additional wheels beyond the minimum of 3, extra wheels in groups of three provide the additional torque capability of a single wheel in each axis. Using less than an additional three wheels is estimated by assuming the wheel provided a third of its torque capability in each primary axis.

$$T_{cap} = \frac{N}{3} I_s \omega_{max} \omega_{gimb} - \max[T_{stab} + T_{ext} + \max(T_{en}), T_{sl}] \quad (B.99)$$

$$\omega_{gimb} = 0.225 \text{ rad/s} \quad (B.100)$$

TACA With a traditional attitude control approach, the ACS provides torques either by changing reaction wheel speeds or by using CMGs. Torque ranges are the same as listed above, 0.02 Nm to 500 Nm, depending on the system used. We assume off-the-shelf hardware for TACA, and the torque capability is known. Flywheels are counter-rotated, and energy is removed at the same rate from both flywheels. Therefore, the net change in angular momentum is zero and there is no torque on the spacecraft. In actuality, there is a small torque due to energy storage which depends on how accurate the flywheel speeds are controlled.

$$T_{cap} = T_{taca} - \max[T_{stab} + T_{ext} + Res \max(T_{en}), T_{sl}] \quad (B.101)$$

3. Momentum Storage

This value in the first estimate provides a rough measure of how well the spacecraft can handle cyclic external torques. It also provides a rough estimate on how frequently reducing excess momentum due to secular external torques is required.

$$\text{Maximum Momentum Storage} = \left\{ \begin{array}{ll} MWA & I_s(\omega_{max} - \omega_{min}) \\ GFA & I_s \omega_{max} \omega_{gimbalrate} \\ TACA & \text{hardware characteristic} \end{array} \right\} \quad (B.102)$$

The secular external torques depend on the orientation of the spacecraft. Larson (16) describes the relationship between orientation and type of torque, either secular or cyclic. Gravity gradient and aerodynamic forces produce secular torques, while solar radiation and magnetic torques are cyclic for Earth-pointing satellites. For an

inertially pointing vehicle, solar radiation and aerodynamics forces produce secular torques, while gravity and magnetic torques are cyclic. The secular torques cause unwanted momentum storage that must eventually be reduced using magnetic torquers or thrusters. The cyclic torques produce the maximum disturbance in a quarter of an orbit. Excess momentum storage provides margin for larger than expected torques and buildup of secular torques.

These torques cause a problem for the momentum wheel approach. Momentum buildup in the MWA is directly linked with energy storage. The added momentum causes a single wheel to reach maximum speed before the other wheels. The ACS halts the addition of energy or momentum to this wheel. Since the other wheels have not reached their maximum speed, more energy can be stored in these wheels. However, if storing more energy would cause the single wheel to spin above its maximum speed, then the total amount of stored energy has reached a maximum. This results in unavailable energy storage. Similarly, a wheel may reach its minimum speed before all the other wheels, further reducing energy storage capacity.

Larson (16) develops the equations to calculate the maximum momentum a spacecraft can store due to external torques. Equation B.112 calculates the worst-case energy storage capability decrease caused by excess momentum storage. The development assumes external torques are such that three of the flywheels did not store any momentum. If the minimum number of wheels cannot provide sufficient momentum storage, additional wheels are added. These wheels are assumed for the purposes of calculations to be used strictly for attitude control and not for energy storage.

$$\text{cyclic} \quad \Delta H_{max} = \int g dt = g \frac{T_{Orb}}{4} (0.637) \quad (\text{B.103})$$

$$\text{secular} \quad \Delta H_{max} = \int g dt = g T_{Orb} \quad (\text{B.104})$$

$$\text{slews} \quad \Delta H_{max} = \int g dt = g_{slew} t_{slew} / 2 \quad (\text{B.105})$$

$$\Delta H_{total} = \sum \Delta H_{max} \quad (\text{B.106})$$

$$\Delta \omega = \frac{M_N \Delta H_{total}}{I} \quad (\text{B.107})$$

$$\text{where } M_N = \begin{cases} 1 & \text{if } N = 4 \\ \frac{\sqrt{2}}{2} & N = 5 \\ \frac{\sqrt{3}}{3} & N = 6 \\ 0.5 & N = 8 \end{cases} \quad (\text{B.108})$$

$$E_{avail \max} = NI(\omega_{\max}^2 - \omega_{\min}^2)/2 \quad (\text{B.109})$$

$$E_{avail \min} = (N - 3)I(\omega_{\max}^2 - \omega_{\min}^2)/2 + \frac{3}{2}I[(\omega_{\max} - \Delta\omega)^2 - (\omega_{\min} + \Delta\omega)^2] \quad (\text{B.110})$$

$$\Delta E_{lost} = E_{avail \max} - E_{avail \min} \quad (\text{B.111})$$

$$\Delta E_{lost} = 3M_N \Delta H_{total}(\omega_{\max} + \omega_{\min}) \quad (\text{B.112})$$

The dedicated ACS flywheels are added in groups of three. Therefore, each group reduces the lost energy in the other flywheels by reducing the momentum storage in energy storing wheels.

4. Mass

The mass of the attitude control system is the mass of additional components required to perform attitude control. We assume attitude determination components would be the same regardless of the attitude control required. Also, we assume any impact on the computer system is identical for all alternatives and can be ignored.

MWA All the components of the attitude control system are also required for energy storage; therefore the mass of attitude control is zero.

GFA The flywheels are also required for energy storage; mass of the flywheels is only counted as energy storage. Only the mass of the gimbaling mechanism counts towards the attitude control system. A gimbaling mechanism is required for each flywheel. The equation to estimate this value is

$$\text{GFA mass per gimbal} = 3\text{kg} + 15\text{kg/m}^3 \times \text{FlyVolume} \quad (\text{B.113})$$

TACA A complete attitude control subsystem is required; hence the mass is that of the system. TACA mass is modeled using actual hardware specifications.

5. Recurring Costs

The costs considered are the costs of additional components.

MWA No new components are necessary; the added cost is \$0.

GFA Added costs are derived from strengthening the magnetic bearings and adding either a tilt table or a gimbal device. The equation used to estimate this additional cost is roughly approximated with the following equation modified from Space and Missile Systems Center's CER (25):

$$\text{Cost per flywheel} = \$30000 (Mass_{gimbal} N)^{0.62} DesignLife^{0.473} \quad (B.114)$$

TACA TACA cost is the cost of the hardware being used.

6. Non-recurring Costs

The costs considered are the costs of hardware RDT&E.

MWA This method has not been used in space yet. RDT&E costs are roughly approximated with the following equation modified from Space and Missile Systems Center's CER (25):

$$\text{Non-recurring Costs} = \$115,000 (Mass_{flywheel} N)^{0.5} \quad (B.115)$$

GFA Gimbale flywheels are still in development with on-going research into the magnetic bearings necessary to provide the requisite torques. The model uses the following modified CER from Space and Missile Systems Center (25) to roughly approximate the non-recurring costs to develop the technology and integrate into a spacecraft.

$$\text{Non-recurring Costs} = \$126,000 Mass_{gimbal}^{0.733} N^{0.5} \quad (B.116)$$

TACA TACA cost is the cost of the hardware being used.

7. Volume

Volume is the additional volume required by the attitude control subsystem.

MWA This method requires no additional volume.

GFA This method requires additional volume beyond the flywheel volume. Volume is required not only for the gimbal mechanism but also to allow the flywheel to rotate. We chose to estimate this additional volume as the following:

$$\text{Additional volume per flywheel} = \text{Volume of the flywheel} \quad (\text{B.117})$$

TACA Volume is the volume of the hardware being used.

8. Reliability

There are two components to reliability. The first is the effect flywheels have on reliability. The second is the effect of everything other than the flywheels. This section covers reliability of attitude control for the three methods. The only effects considered are those of flywheel failures and actuator failures. Failures in computers, attitude determination, thrusters, or magnetic torquers are not considered.

MWA Reliability for the MWA is the reliability of the flywheels. The minimum number of flywheels required to perform attitude control and simultaneous energy storage is 4. When the number of working flywheels drops to 3 or less, or the remaining flywheels are coplanar, the flywheels cannot perform both ACS and ESS functions simultaneously. As long as 4 non-coplanar flywheels are working, and all non-functioning flywheels are decreased to zero speed, both attitude control and energy storage will be functional.

Possible effects of non-fatal loss of flywheels:

- Attitude control is no longer optimized
- Possible decrease in stability
- No longer using best sets of flywheels to provide specific torques
- Increase in power required to counter some torques

- Limited range of operation for maximum and minimum wheel speed
- Storing and removing energy from a flywheel uses a torque that must be countered by other flywheels. If a flywheel is lost, other flywheels must cover any torques it previously countered. If there are many more than 4 flywheels, this is not a problem. With a decrease from 5 to 4 flywheels, there is a large chance that at least one flywheel will have to reduce the range of speeds within which it operates.

GFA Reliability for the GFA consists of two parts. The first part is the reliability of the flywheels. The second part is the reliability of the actuators. Under the GFA, the flywheels are operated strictly for energy storage. The actuators provide the torque capability for attitude control. If a flywheel fails, the actuator is no longer useful because the flywheel speed is assumed to be zero. The actuator cannot provide torque using a stationary flywheel. One full flywheel is lost to attitude control. If the actuator fails, the assumption is that the actuator locks and can no longer provide torque. The flywheel can still be used for energy storage. Beginning with 3 flywheels, all with working actuators, if a flywheel fails, 2 flywheels with actuators remain. This is 4 degrees of freedom using the remaining flywheels in a differential rate mode similar to the MWA. Full attitude control would be possible. Energy storage would be severely limited or not possible at all. Beginning with 3 or more flywheels, as long as three working flywheel-actuator pairs are still working, full attitude control is possible at a lesser degree of stability.

TACS Reliability for TACS is available for most commercially available systems. Typical TACS use 4 flywheels, three orthogonal and one skewed. As long as three flywheels are working, full attitude control is possible.

B.6 Modeling and Optimization

B.6.1 System-Level Modeling. Once the functions were represented with equations, we collected the lower-level function models into a system-level model.

The system-level model combines the appropriate functions necessary to form our defined system. The system-level model imposes system performance requirements and trades on the data computed by the lower-level models. It is responsible for selecting, configuring, sizing, scoring and reporting the properties of a specific alternative design.

B.6.1.1 General Algorithm. The system-level model executes the following algorithm:

1. Pick a design alternative (a specific set of alterables)
2. Verify that the alternative is allowed
3. Verify that the alternative is desired
4. Configure the alternative as a system
5. Size the system to meet ACS requirements first, then to meet ESS requirements, and finally to meet the reliability requirements, if necessary
6. Compute the system's performance
7. Compute utilities from the performance measures
8. Compute the objective measures from the utilities
9. Compute the value score from the objective measures
10. Report the system characteristics and scores
11. Iterate until all alternatives have been examined

This section explains the details of the system-level model.

B.6.1.2 Alternative Selection and Configuration. The system-level model uses an exhaustive loop system to select alternative designs. There are four loops in the model, one for each alternative. They are:

ACS Scheme The attitude control scheme has eight possible values. The attitude control scheme is either IPACS/MWA, IPACS/GFA, or one of the six TACS in the database.

ESS Scheme The energy storage scheme has 12 possible values. These correspond to the six flywheel and six battery data sets in the database.

PMAD Scheme The power management and distribution scheme has two possible values. They identify whether the system is a peak power tracking or direct energy transfer design.

Launch Scheme The launch scheme has two possible values. The 'locked' value indicates the flywheels are not used until after the solar arrays are deployed. This system requires a primary power supply that it uses until the solar array is deployed and the flywheels are spun up. The 'spinning' value indicates that the flywheels provide primary power during the launch-to-deployment phase of the mission. This option requires a robust design that can operate in the launch environment.

The model considers, at most, 384 alternatives. Of course some of the possible alternatives do not make sense. For example, batteries combined with a spinning launch scheme is an infeasible combination. Similarly, IPACS are incompatible with batteries. The configuration method in the system model is used to determine which systems will actually be configured and which systems will be discarded without analysis.

The configuration method eliminates alternatives based on two criteria:

1. Is the combination of alterables in the alternative compatible?
2. Are any of the alterables in the alternative undesirable?

The first question is used to eliminate those alternatives that aren't logical. The second question screens out alternatives that include hardware from the database that was not selected for a particular analysis (only three of the six flywheels in the database may be required for a particular study, etc.). Once an alternative has passed these checks, it is used to activate the correct hardware models and to configure the components of the system-level model. The system model can now start the iterative sizing process.

B.6.1.3 Sizing. The sizing process is based on three sets of requirements.

They are:

1. ACS requirements

- (a) Absolute minimum number of wheels to perform ACS functions
 - (b) Torque requirements
 - (c) Momentum storage requirements
2. Power requirements
 3. Reliability requirement

The system-level model sizes to meet these requirements, starting with ACS and working down to reliability. Specific implementations are discussed below.

Meeting ACS Requirements. An initial sizing attempt is made based on the number of flywheels the attitude control system requests. The attitude control system initial guess is based primarily on the number of wheels required to perform attitude control functions. For non-IPACS designs, this is two flywheels. For IPACS/GFA designs, it is at least three; for IPACS/MWA, four. It has no bearing on TACS/B systems. IPACS designs are also required to meet torque and momentum requirements. The system model sizes to meet these requirements by adding wheels, one at a time, until the requirements are met. TACS may or may not meet the requirement. It is assumed that if TACS are to be compared to IPACS, the analyst would put mission-compatible TACS hardware into the database.

Meeting Power Requirements. System power requirements are based on the power load, selected hardware, and mission profile. The energy storage system is sized to meet the energy storage required to operate all systems for the maximum eclipse duration. Equation B.118 is used to determine the total energy storage requirement.

$$E_{required} = E_{max\ eclipse} + E_{ESS\ housekeeping} + E_{ACS\ requirement} \quad (B.118)$$

The number of units required to meet the energy storage need is given by Equation B.119.

$$n_{ES\ Units} = \frac{E_{required}}{Unit\ Capacity} \quad (B.119)$$

This number must be an integer, so some adjustment is also required. To make the result of Equation B.119 an integer the answer is first truncated. The resulting integer is incremented by one for an IPACS, by two for a TACS/FES system and by one for a TACS/B system. If this final result is larger than the number of wheels required to meet the ACS requirements, it becomes the new minimum number of wheels required for the system.

Meeting Reliability Requirements. Sizing for reliability is accomplished by evaluating the system reliability equation and comparing the result to the reliability requirement. If the system reliability is less than the requirement, additional flywheels are added until the reliability requirement is met or until the system reliability stops improving. Reliability sizing is not performed on systems that use TACS. TACS are ignored because the reliability sizing algorithm would increase ESS reliability to senseless levels to compensate for the fixed TACS reliability. This would tend to bias against conventional attitude control systems and produce an unrealistic trade.

B.6.1.4 Performance. After the system is sized to meet requirements, the system performance measures need to be calculated. These performance measures are:

1. **Energy Storage Performance** This is the ratio of energy storage capacity to the energy storage required. A value of 1.0 indicates that the energy storage system meets the given requirement.
2. **Attitude Control Performance** This performance measure is a weighted sum of the torque capability and the momentum storage capacity.
3. **Cost** This measure is the sum of the non-recurring and recurring costs for the alternative design.
4. **Mass** This measure is the ratio of the Power and Attitude Control Subsystems (PACS) mass to the system dry mass.
5. **Volume** This measure is the total PACS volume
6. **Reliability** This measure is the overall reliability of the PACS

Detailed methods are presented next.

Energy Storage Performance. Energy storage performance is computed using Equation B.120.

$$P_{Energy\ Storage} = \frac{n_{ES\ Units} \times Unit\ Capacity}{E_{Required}} \quad (B.120)$$

The number of energy storage units ($n_{ES\ Units}$) is driven by ACS, ESS and reliability requirements. This causes the range for $P_{Energy\ Storage}$ to be greater than or equal to 1.0.

Attitude Control Performance. Attitude Control performance is calculated using Equation B.121.

$$P_{Attitude\ Control} = 0.25 \times (Momentum\ Capacity) + 0.75 \times (Torque\ Capacity) \quad (B.121)$$

This composite performance measure for attitude control has no physical meaning and is used only to discriminate between alternative designs. The underlying properties, torque and momentum storage capacity, are calculated in the hardware model for attitude control.

Cost. The system cost is given by Equation B.122.

$$P_{Cost} = \sum Subsystem\ Cost_{non-recurring} + n_{manufactured} * \sum Subsystem\ Cost_{recurring} \quad (B.122)$$

The subsystem recurring and non-recurring costs are calculated in the hardware component models and reported in dollars inflated or deflated to a specified base year (16). The current model supports fiscal years 1992 through 2005.

Mass. The system mass is calculated using Equation B.123.

$$P_{Mass} = \sum Subsystem\ Mass \quad (B.123)$$

The subsystem masses are calculated in the hardware models.

Volume. The system volume is calculated using Equation B.124.

$$P_{Volume} = \sum Subsystem\ Volume \quad (B.124)$$

The subsystem volumes are calculated in the hardware models.

Reliability. The system reliability is determined by using Equation B.125

$$R_{system} = R_{ESS} \times R_{ACS} \times R_{PMAD} \times R_{EGS} \times R_{catastrophe} \quad (B.125)$$

The terms in Equation B.125 are dependent on the particular system configuration. R_{PMAD} is the reported end-of-life reliability for the PMAD system. R_{EGS} is the reported end-of-life reliability for the solar array. $R_{catastrophe}$ is the probability of avoiding catastrophic flywheel failure resulting in the loss of the satellite. R_{ESS} is a k out of n reliability problem, where k is the minimum number of energy storage units required to meet the energy storage need and n is the total number of energy storage units on board. In the case of a TACS, this reduces to a simple series reliability calculation ($k = n$). For IPACS-type systems it is possible that there may be some degree of redundancy ($k \neq n$). Finally, R_{ACS} is either a simple end-of-life reliability (TACS) or a k out of n system (IPACS). Equation B.126 is the general form of the k out of n .

$$R = \sum_{i=k}^n \binom{n}{i} p^n (1-p)^{n-i} \quad (B.126)$$

The catastrophic failure model is based on whether or not the system contains flywheels as components. The risk of catastrophic failure is proportional to the number of flywheels in the system.

$$R_{catastrophe} = 1.0000 \quad (B.127)$$

$$R_{catastrophe} = (0.9999)^{n_{flywheels}} \quad (B.128)$$

Equation B.127 is used when flywheels are not in the alternative being evaluated, while Equation B.128 applies in all other cases.

The performance measures computed with the preceding equations are used to calculate the objective measures. These are discussed next.

B.6.1.5 Utility. The objective measures are computed by transforming the performance measures using a utility equation. As implemented in this model, utility is a value on a zero to one scale which indicates how much a decision-maker values a particular level of performance. The decision-maker's value increases with increases in performance up to a certain level, beyond which he is indifferent to further improvement. This level of performance receives a utility of one; higher performance scores receive the same utility score. At the other end of the spectrum is the point of minimum utility. This is the point at which the performance is minimally useful to the decision-maker and scores a zero; lower performance will also score zero utility.

Utility profiles are required for each of the performance measures. These profiles specify the points where utility is maximized or minimized, as well as several intermediate points. The model interpolates between these points to compute the utility scores for a particular performance measure. The utility scores become the objective measures (Equation B.129) that are used to compute the value score of the alternative.

$$\bar{O} = U(\bar{P}) \quad (\text{B.129})$$

where \bar{O} is the vector of objective measures and \bar{P} is the vector of performance measures. U is the utility transformation function.

B.6.1.6 Value. The value score is the overall ranking of the system. It is based on the weights provided by the decision-maker and the computed objective measures. It is given by Equation B.130.

$$V = \bar{w}^T \bar{O} \quad (\text{B.130})$$

where \bar{w} is a vector of the decision weights and \bar{O} is the vector of objective measure values for the system being evaluated.

B.6.1.7 Optimization. The optimal design is selected by sorting the list of alternate designs according to their reported value score. The model does not currently optimize individual designs for value; it merely sizes alternative systems to meet or exceed all specified requirements. Specific design optimization (internally optimizing the optimal design to squeeze the highest possible value score from it) could be accomplished by manually changing the number of flywheels and observing the change in the value score to determine its peak value. For the purposes of our study we are limiting optimization by determining which alternative best meets the design requirements and by performing sensitivity analysis on the optimal result to determine the robustness of the optimal design.

B.7 Decision-Making

The purpose of the Decision-Making Step is to examine the results of the *Modeling and Optimization* step and to either select an alternative design for implementation or select one or more alternatives for a more detailed analysis. The objective of our design study is to be able to recommend which type and configuration of power and energy storage systems should be used for a given satellite mission. The basic questions we seek to answer are:

- Should flywheels replace batteries?
- Should flywheels replace batteries and traditional attitude control systems in a combined system (IPACS)?
- If flywheels can be applied in a FES system or IPACS, when should they? Should they be used only for FES or as an IPACS?
- For an IPACS, should the flywheels be CMGs or momentum wheels for attitude control?
- Should flywheels be spinning at launch or should they be locked?
- What flywheel characteristics have the most effect on the performance of flywheel-based systems?
- What are areas of significance for future research?

To answer these questions we defined the problem; our needs, alterables, and constraints; and our value system. After synthesizing the alternative designs we built a model of the system. The model built in the *Systems Analysis* and *Modeling and Optimization* steps underwent continuous refinement until we were satisfied with the fidelity of the model. We then conducted the Design Space Exploration, Point Analysis, Variability Analysis, and Sensitivity Analysis with our model. The results we obtained were of sufficient fidelity to allow us to answer the questions posed in our design study objective.

Our decision is that this iteration of the design study provides sufficient detail and satisfies our objectives. The results were selected for implementation and no further iterations are required.

B.8 Planning for Action

The results of the Decision-Making step of the second iteration indicate that there is no need for another phase of the project.

B.8.0.8 Problem Definition. In the *Problem Definition* step, the goals were to attain a full understanding of the problem, and to define all relevant problem elements. The restatement of our 9 products of problem definition, as recommended in the *Planning for Action* step of the first iteration, lead to the clear delineation of all requisite aspects of the problem within the specified scope of the project.

B.8.0.9 Value System Design. The goals of the *Value System Design* step were to define the objectives of the system study, define the objective measures of the stated objectives, and assign subjective weights to the stated objective measures. Earlier in this second iteration, the objective measures were defined at the higher level required. Also, information that lead to the development of applicable weights was acquisitioned. The finalized objective measures and weights were completed in this step. The system model created was then closer to producing viable data for analysis.

B.8.0.10 System Synthesis. In the *System Synthesis* step, the goals were to conceptualize alternative approaches for obtaining the objectives and to describe each alternative approach. The second iteration produced feasible alternatives that were described in detail. No further activity to advance this area is required.

B.8.0.11 System Analysis. The goal of the *System Analysis* step was to evaluate the properties of the elements performing the allocated functions that determine the performance measures. Formulation of a scoring method based on the performance measures was completed during the second iteration in order to determine the degree to which the alternatives satisfy the design requirements.

B.8.0.12 Modeling and Optimization. In the *Modeling and Optimization* step, the goal was to model the alternatives generated in the *System Synthesis* step and determine how they perform with respect to the value system we have defined. In the second iteration, a more detailed breakdown of the system was achieved. The definition of the final system model was completed, and the long awaited full optimization procedure occurred.

B.8.0.13 Systems Engineering Process. Hall's model is most appropriately applied to full-scale design efforts that are not time constrained to the degree that this development endeavor was. Having met with the necessity of reevaluating our process due to the pressures of the design environment, we decided to take an approach that sought to coalesce the best features of Hall's Model with Integrated Modeling. This is discussed in detail in Chapter 2.

Appendix C. Annotated References

Keyword: bs_oglevie01
Reference Type: techreport
Author(R): Oglevie, Ronald E. and David B. Eisenhaure
Title(R): Advanced Integrated Power and Attitude Control System (IPACS) Study
Institution(R): NASA Contractor Report 3912
Year(R): 1985
Editor:
Organization:
Publisher:
Address:

Annotation:

OVERVIEW:

This study is projected toward the definition of an IPACS for the international space station. Although the direction of application is outside the scope of our endeavor, there are aspects of the design process which we may find useful.

11-28: IPACS Configuration Definition - Information on System Requirements Definition, Redundancy, Component- Level Requirements

29-50: Space Station System-Level Trades - Information on Trade Data Development and Life Cycle Cost Trades

87: Appendix C - Magnetic Bearing Design Analysis

119: Appendix D - Power Conversion Design Analysis

135: Appendix E - Integrated Component Design Trades

Keyword: bs_eisenhaure02
Reference Type: techreport
Author(R): Eisenhaure, David B., et al.
Title(R): A Combined Attitude, Reference, and Energy Storage System for Satellite Applications
Institution(R): AIAA 22nd Aerospace Sciences Meeting
Year(R): 1984
Editor:
Organization:
Publisher:
Address:

Annotation:

OVERVIEW:

This paper discusses a technique for combining the functions of energy storage, attitude control, and attitude reference systems.

1: Figure 1 - The impact of efficiency and energy density on satellite power system mass

2: Battery information

Figure 2 - Battery cell-life as a function of energy density

Flywheel information

Tables 2 and 3 - Characteristics of several flywheels sized for a typical LEO mission

3: Comparison of NiCad, NiH₂, and Flywheels with a brief attitude control discussion

3-4: Attitude rate sensing discussion

4-10: Breakdown of Charles Stark Draper Labs Satellite CARES System

Keyword: bs_odea03
Reference Type: techreport
Author(R): O'Dea, Stephen, et al.
Title(R): Design and Development of a High Efficiency Effector for the Control of Attitude and Power in Space Systems
Institution(R): Proceedings of the 20th Annual Intersociety Energy Conversion Engineers Conference vol. 2
Year(R): 1985
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

This paper outlines the design and development of an advanced CARES system.

Requirements are discussed in order to arrive at CARES specifications for the international space station and the Solar Max Mission version of the Multi-Mission Modular Spacecraft (SMM/MMS).

Discussion of Components:

- flywheel/spoke system
- large-angle magnetic suspension
- motor/generator
- motor/generator electronics

System Integration

Keyword: bs_flanagan04
Reference Type: techreport
Author(R): Flanagan, R.C.
Title(R): Design, Manufacture and Test Results for Four High Density Fibre Composite Rotors
Institution(R): Proceedings of the 21st Annual Intersociety Energy Conversion Engineers Conference vol. 2
Year(R): 1986
Editor:
Organization:
Publisher:
Address:

Annotation:

OVERVIEW:

Design and test results for four high energy density fibre-composite rotors.

902-903: Rotor Stress Profiles

904: Rotor Testing and Dynamic Behavior

906: Rotor Dynamic Behavior

Keyword: bs_niemeyer05
Reference Type: techreport
Author(R): Niemeyer, W. L., et al.
Title(R): A High Efficiency Motor/Generator for Magnetically Suspended Flywheel
Energy Storage System
Institution(R): Proceedings of the 24th Annual Intersociety Energy Conversion Engineers Conference vol. 3
Year(R): 1989
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

The research presented in this paper discusses the theory and design of a brushless direct current motor for use in a flywheel energy storage system.

Based on information from Niemeyer's Masters Thesis.

Keyword: bs_niemeyer06
Reference Type: mastersthesis
Author(R): Niemeyer, William Leland
Title(R): Design of a High Efficiency Motor for Flywheel Energy Storage
School(R): University of Maryland
Year(R): 1988
Address:
Month:
:
:

Annotation:

OVERVIEW:

Detailed development concerning the optimization of the brushless DC motor/generator for use with flywheel energy storage. Most of the information is beyond the scope of our study, but some may be useful.

Keyword: bs_notti07
Reference Type: techreport
Author(R): Notti, J.E., A. Cormack III, and W.C. Schmill
Title(R): Integrated Power/Attitude Control System (IPACS) Study Vol. 1 Feasibility Studies
Institution(R): NASA CR 2383
Year(R): 1974
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

Volume 1 contains the trade-off studies performed to establish the feasibility, cost effectiveness, required level of development, and boundaries of application of IPACS to a wide variety of spacecraft.

Keyword: bs_notti08
Reference Type: techreport
Author(R): Notti, J.E., et al.
Title(R): Integrated Power/Attitude Control System (IPACS) Study Vol. 2 Conceptual Designs
Institution(R): NASA CR 2384
Year(R): 1974
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

Volume II presents the conceptual designs for a free flying research application module (RAM), and for a tracking and data relay satellite (TDRS). Volume II also contains results from dynamic analyses and simulations of the IPACS conceptual designs.

Keyword: bs_winter09
Reference Type: techreport
Author(R): Winter, Christopher P. and Vincent L. Teofilo
Title(R): Electrical Power Trade Analyses for Space Surveillance Satellites
Institution(R): Proceedings of the 24th Annual Intersociety Energy Conversion Engineers Conference vol. 1
Year(R): 1989
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Introduction:

This paper describes the results of the trade studies performed to select the power source for a typical surveillance satellite, as well as for compatible power-conditioning equipment, power control and distribution architecture, and energy storage components.

Keyword: bs_thornton10
Reference Type: techreport
Author(R): Thornton, R. D.
Title(R): Power Management and Distribution Equipment Development for Space Applications
Institution(R): Proceedings of the 26th Annual Intersociety Energy Conversion Engineers Conference vol. 2
Year(R): 1991
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

Space Applications have consistently advanced the boundaries of many areas of technology including electric power systems. As the technology has advanced, power and distribution equipment for space applications has grown to include a broad spectrum of switchgear and conversion devices. Space Station Freedom has continued this tradition of progress and some of the current developments in the area of dc power switching and conversion equipment are presented in this paper.

Keyword: bs_ashley11
Reference Type: techreport
Author(R): Ashley, Christopher R., et al.
Title(R): An Electrical Power System for a Mobile Communications Satellite
Institution(R): Proceedings of the 27th Annual Intersociety Energy Conversion Engineers Conference vol. 1
Year(R): 1992
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

A regulated high voltage solar array/battery system using large area silicon solar cells and a group of Ni-H energy storage cells meets the requirements of a multikilowatt geosynchronous orbit application. The electrical power bus operates at approximately 50 Vdc, minimizing unbalanced torques and distribution harness weight. The program design goal of 12 years of operation on station, including up to 2 years of orbital storage, represents a significant extension of total mission life.

Keyword: bs_yousef12
Reference Type: techreport
Author(R): Yousef, Hassan
Title(R): Power Management and Distribution for Future Spacecraft
Institution(R): Proceedings of the 24th Annual Intersociety Energy Conversion Engineers Conference vol. 1
Year(R): 1989
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

This report describes the current and future spacecraft power system, including its four major subsystems consisting of energy generation, energy storage, energy management, energy conditioning, and energy control/distribution subsystem. Additionally, this report will illustrate the power requirements for current and future spacecraft.

The on-going increase in spacecraft power requirements is under study and concepts of efficient, autonomous, and environmentally protected power systems will be addressed in this report.

Finally, spacecraft power system modularization (Figures 1 and 2) and standardization for the same class of spacecraft, such as communications and sensor spacecraft, will be addressed in this report.

Keyword: bs_kenny13
Reference Type: techreport
Author(R): Kenny, Barbara H.
Title(R): An Analysis of Space Power System Masses
Institution(R): Proceedings of the 25th Annual Intersociety Energy Conversion Engineers Conference vol. 1
Year(R): 1990
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

This paper analyzes various space electrical power system masses with emphasis on the power management and distribution (PMAD) portion. The electrical power system (EPS) is divided into functional blocks: source, interconnection, storage, transmission, distribution, system control and load. The PMAD subsystem is defined as all the blocks between the source, storage and load, plus the power conditioning equipment required for the source, storage and load. The EPS mass of a wide range of spacecraft is then classified as source, storage or PMAD and tabulated in a database. The intent of the database is to serve as a reference source for PMAD masses of existing and in-design spacecraft.

The PMAD masses in the database range from 40 kg/kW to 183 kg/kW across the spacecraft systems studied. Factors influencing the power system mass are identified.

These include the total spacecraft power requirements, total amount of load capacity and physical size of the spacecraft. It is found a new "utility" class of power systems, represented by Space Station Freedom, is evolving.

Keyword: bs_kimnach14
Reference Type: techreport
Author(R): Kimnach, Greg L.
Title(R): Design of a Power Management and Distribution System for a Thermionic
-Diode Powered Spacecraft
Institution(R): Proceedings of the 31st Annual Intersociety Energy Conversion Engineer
rs Conference vol. 1
Year(R): 1996
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

The Electrical Systems Development Branch of the Power Technology Division at the NASA Lewis Research Center in Cleveland, Ohio is designing a Power Management and Distribution (PMAD) System for the Air Force's Integrated Solar Upper Stage (ISUS) Engine Ground Test Demonstration (EGD).

The ISUS power system is to provide 1000We at 28+/-6Vdc to the payload/spacecraft from a maximum Thermionic Diode (TID) generation capability of 1070We at 2200K.

Producing power with this quality, protecting the spacecraft from electrical faults and accomodating operational constraints of the TIDs is the responsibility of the PMAD system. The design strategy and system options examined along with the proposed designs for the Flight and EGD configurations are discussed herein.

Keyword: bs_suich15
Reference Type: techreport
Author(R): Suich, Ronald C., and Richard L. Patterson
Title(R): Balancing Reliability and Cost to Choose the Best Power Subsystem
Institution(R): NASA Technical Memorandum 104453
Year(R): 1991
Editor:
Organization:
Publisher:
Address:

Annotation:

Abstract:

How does a design engineer or manager choose between a power subsystem with .990 reliability and a more costly power subsystem with .995 reliability? When is the increased cost of a more reliable power subsystem justified?

A mathematical model is presented for computing total (spacecraft) subsystem cost including both the basic subsystem cost and the expected cost due to the failure of the subsystem. This model is then used to determine power subsystem cost as a function of reliability and redundancy. Minimum cost and maximum reliability and/or redundancy are not generally equivalent. Two example cases are presented. One is a small satellite, and the other is an interplanetary spacecraft.

Keyword: bs_chetty16
Reference Type: techreport
Author(R): Chetty, P. R. K., et al.
Title(R): TOPEX Electrical Power System
Institution(R): Proceedings of the 26th Annual Intersociety Energy Conversion Engineers Conference vol. 1
Year(R): 1991
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

This paper presents the TOPEX mission requirements which impact the power requirements and analyses. A description of the EPS including energy management and battery charging methods that were conceived and developed to meet the identified satellite requirements is included.

Keyword: bs_morgan17
Reference Type: techreport
Author(R): Morgan, W. T., et al.
Title(R): AXAF Electrical Power System
Institution(R): Proceedings of the 24th Annual Intersociety Energy Conversion Engineers Conference vol. 1
Year(R): 1989
Editor:
Organization:
Publisher:
Address:

Annotation:

OVERVIEW:

This paper presents the Electrical Power Subsystem (EPS) requirements for the Advanced Astrophysics Facility (AXAF).

The EPS topology is delineated, and a subsystem electrical block diagram is presented.

The EPS sizing procedure is outlined, followed by a brief discussion of system reliability.

Keyword: bs_krauthamer18
Reference Type: techreport
Author(R): Krauthamer, S.
Title(R): The State of the Art of dc Power Distribution Systems/Components for Space Applications
Institution(R): NASA CR-182743
Year(R): 1988
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

This report is a survey of the state of the art of high voltage dc systems and components. This information can be used for consideration of an alternative secondary distribution (120 Vdc) system for the Space Station. All HVdc components have been prototyped or developed for terrestrial, aircraft, and space applications, and are applicable for space application with appropriate modification and qualification. HVdc systems offer a safe, reliable, low mass, high efficiency and low EMI alternative for Space Station secondary distribution.

Keyword: bs_chetty19
Reference Type: book
Author(s) -or-: Chetty, P.R.K.
Editor(s) (R): Tab Professional and Reference Books
Title(R): Satellite Technology and It's Applications
Publisher(R): Tab Professional and Reference Books
Year(R): 1991
Address:
Edition:
Month:

Annotation:

From the Introduction:

This book contains two parts. Part 1 deals with satellite building blocks, and Part 2 deals with the various applications. Part 1 details the state of the art technology of various building blocks that go into making the satellite after an introduction to satellites, and Part 2 discusses the various applications of satellites like telecommunications, broadcasting, weather forecasting (meterology), predicting crop yields, mineral exploration, land survey, status of forestry, growth of ultra-clean crystals for faster computers, and growth of products for making new and cheaper medicines.

Keyword: bs_reeves20
Reference Type: techreport
Author(R): Reeves, Emery I.
Title(R): FLTSATCOM Case Study in Spacecraft Design
Institution(R): AIAA Professional Studies Series
Year(R): 1979
Editor:
Organization:
Publisher:
Address:

Annotation:

OVERVIEW:

This report outlines the FLTSATCOM development program in summary form.

Keyword: bs_cheung21
Reference Type: mastersthesis
Author(R): Cheung, Sing Nan
Title(R): Minimum Power Design of a Magnetic Bearing
School(R): University of Arizona
Year(R): 1974
Address:
Month:
:
:
:

Annotation:

From the Abstract:

This thesis examines a novel controller (VZP system) which overcomes the destabilizing magnetic force in the axial direction using an electromagnetic actuator and a rate coil to sense the axial velocity. The steady state power required to overcome an external force is nearly zero.

Keyword: bs_chiao22
Reference Type: techreport
Author(R): Chiao, T. T.
Title(R): Fiber Composite Materials Development for Flywheel Applications
Institution(R): 1980 Flywheel Technology Symposium
Year(R): 1980
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

The fiber composite materials program was formulated to support the development of various flywheel rotor designs for energy storage. Our areas of activity include matrix and fiber evaluations, static properties of composites for engineering design, and lifetime data to predict the long-term performance of composites. In this article we summarize the highlights of our technical accomplishments to date, and describe our overall program and the key technical issues involved for future activities.

Keyword: bs_hickey23
Reference Type: techreport
Author(R): Hickey, J. S. and D. L. Kerr
Title(R): Evaluation of Flywheel Containment
Institution(R): Supplement to the 1980 Flywheel Technology Symposium
Year(R): 1980
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

This paper summarizes a program which was undertaken to provide a sound basis for defining the material, configuration, and dimensions of the containment structure to be used in the Flywheel Energy Storage System being developed for a transit bus by the General Electric Company under Contrant DOT-TSC-1670 with the Transportation Systems Center of the U.S. Department of Transportation. This paper includes a brief description of the full size system which utilizes a multi-disc steel flywheel.

It also describes the failure modes of concern, the procedure used to obtain a preliminary definition of the containment structure required to contain a flywheel rotor failure, and the rationale for deciding to implement a small scale containment test program to give more definitive answers to the containment design problem.

The small scale test program facilities and equipment are described as well as the test program and results up through early April 1980.

Keyword: bs_kulkarni24
Reference Type: techreport
Author(R): Kulkarni, Satish V.
Title(R): The Flywheel Rotor and Containment Technology Development Program of
the U.S. Department of Energy
Institution(R): Supplement to the 1980 Flywheel Technology Symposium
Year(R): 1980
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

The status of the flywheel rotor and containment technology development program in the United States, sponsored by the U.S. Department of Energy, is reviewed in this paper. The specific objectives of the effort are delineated, and prototype composite rotor designs are described. These prototypes are being evaluated to identify promising designs for future development and end-use applications. The composite-laminated-rotor development effort at Lawrence Livermore National Laboratory (LLNL) and General Electric are also discussed, including design modifications to improve energy density, fabrication of thick composite laminates and filament-wound rings, low-cost manufacturing processes, generation of rotor-design data, nondestructive inspection, and spin testing of rotors. Finally, an assessment is made of the current rotor-burst-containment design philosophy and of the applicability of jet-engine fan-blade containment technology to flywheel systems. Based on this assessment, preliminary design criteria for containment are evolved.

Keyword: bs_portnov25
Reference Type: techreport
Author(R): Portnov, G. G., and I. N. Barinov
Title(R): Experimental Study of the Natural Vibration Frequencies and Decrement
s of Composite Rim Flywheels
Institution(R): Mechanics of Composite Materials vol. 28 no.5
Year(R): 1993
Editor:
Organization:
Publisher:
Address:

Annotation:

From Paragraph 1:

We investigated five rim flywheels, which were distinguished one from the other by the number and slope of the spokes relative to the plane of the rim, and by the masses of the rim and hub. The mass and geometric characteristics of the flywheels are presented in Table 1 (1-4 - flywheels with radial spokes; 5 - flywheel with chord spokes). The rim of the flywheel and the spokes were fabricated by winding fiberglass with an elastic modulus of 5.5 GPa and a specific density of 1950 kg/m³; the hub was fabricated from duralumin.

The basic purpose of the study (considering the limited number of flywheels) was to ascertain the principal characteristic features of their dynamic properties, and to assess the possibility of controlling them and of using simplifying hypotheses for theoretical description of flywheel energy-accumulation systems.

Keyword: bs_mokeev26
Reference Type: techreport
Author(R): Mokeev, A. M., and S. G. Sidorin
Title(R): Stress Analysis of Fibrous Flywheels with Binder Taken into Account
Institution(R): Strength of Materials vol. 24 no. 6
Year(R): 1992
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

The article suggests a method of stress analysis of structures made of fibrous materials, based on the net theory of composites and taking into account the effect of rigidity and strength properties of the binder. Comparative evaluations are presented of the stress analysis of rim-spoke flywheels by the net method and with the aid of the described method.

Keyword: bs_portnov27
Reference Type: techreport
Author(R): Portnov, G. G., et al.
Title(R): Energy Capacity of Flywheels with Quasiisotropic or Isotropic Core
Institution(R): Mechanics of Composite Materials vol. 29 no.1
Year(R): 1993
Editor:
Organization:
Publisher:
Address:

Annotation:

From Paragraph 1:

Quasiisotropic disks are made of symmetrically laminated composite in which unidirectional monolayers ($n > 3$) are placed at the angles $\theta_{sub\ i} = \pi \cdot i / n$ ($i = 0, 1, \dots, n-1$) to the selected direction. To ensure joint motion of the rim and disk without radial stresses arising on their contact surfaces during rotation, materials have to be chosen which match each other adequately in elastic properties, and the optimal thickness of the rim and optimal interference fit induced by pressing or cooling to cryogenic temperatures [8] have to be calculated so as to ensure maximal specific mass and volumetric energy capacity. In the publications cited above these problems were dealt with in part only; the present work attempts a more complete analysis.

Keyword: bs_eisenhaure28
Reference Type: techreport
Author(R): Eisenhaure, David, et al.
Title(R): MOSFET Based Power Converters for High-Speed Flywheels
Institution(R): 1980 Flywheel Technology Symposium
Year(R): 1980
Editor:
Organization:
Publisher:
Address:

Annotation:

From the Abstract:

High-speed energy storage flywheels place severe requirements on the design of solid-state power converters, for a flywheel coupled power converter must combine high efficiency over a wide range of power and flywheel speed with relatively low production costs.

Recent developments in high power MOSFET transistors have significantly increased the technical and economical viability of solid-state power converters for flywheel applications. This paper reviews and compares the control requirements for both induction and permanent magnet machines. The characteristics of the Charles Stark Draper Laboratory's baseline drive systems in both these areas are given. The general requirements for a Pulse Width Modulated (PWM) inverter are outlined together with the application advantages of MOSFET devices. The general efficiency characteristics of semiconductor alternatives in motor drive applications are developed in the context of improved economic performance.

Keyword: bs_larson29
Reference Type: book
Author(s) -or-:
Editor(s) (R): Larson, Wiley J. and James R. Wertz
Title(R): Space Mission Analysis and Design
Publisher(R): Microcosm, Inc. and Kluwer Academic Publishers
Year(R): 1996
Address:
Edition:
Month:

Annotation:

Section 10.4.6 was referenced during PMAD model formulation. The equations from Table 10-27 were used for PMAD subcomponent mass estimation.

Keyword: bs_nguyen30
Reference Type: techreport
Author(R): Nguyen, P., et al.
Title(R): Unmanned Space Vehicle Cost Model, Seventh Edition
Institution(R): Space and Missile Systems Center
Year(R): 1994
Editor:
Organization:
Publisher:
Address:

Annotation:

Minimum Unbiased Percentage Error (MUPE) Subsystem and Component Level Cost Estimating Relationships (CER) were referenced during PMAD model formulation. The equations from 6-66 and 6-67 were used for PMAD system nonrecurring and recurring cost estimation.

Keyword: bs_mostert31
Reference Type: book
Author(s) -or-:
Editor(s) (R): Mostert, Robert
Title(R): Flight Hardware Survey Revision No. 1
Publisher(R): Jet Propulsion Laboratory, California Institute of Technology
Year(R): 1993
Address:
Edition:
Month:

Annotation:

From the Purpose:

The purpose of this document is to provide engineers in JPL studies, pre-projects, and flight projects quick access to consistent information on the characteristics and availability of spacecraft systems, subsystems, and components. It is the purpose of the Flight Hardware Survey to collect and maintain a current understanding of the hardware that is available in industry.

This publication was referenced for PMAD subcomponent characteristics.

Keyword: bs_mcnabb32
Reference Type: mastersthesis
Author(R): McNabb, Mark W.
Title(R): An Investigation of Certain U.S. Government "Lessons Learned" Process
es and Their Application to USAF Research, Development, and Acquisition Project Managers
School(R): Air Force Institute of Technology
Year(R): 1991
Address:
Month:
:
:

Annotation:

From the Abstract:

This study investigated the types of processes used by certain U.S. government agencies to provide "lessons learned" that could be used by USAF research, development, and acquisition project managers. Areas of interest include the types of agencies that could provide "lessons learned;" the sources, methods, and techniques these agencies use to process (acquire, maintain, and disseminate) management information such as lessons learned; and the measurement criteria the agencies use to evaluate the effectiveness of these processes.

Referenced for information on Lessons Learned.

Keyword: bs_love33
Reference Type: book
Author(s) -or-: Love,Tom
Editor(s) (R):
Title(R): Object Lessons: Lessons Learned in Object-Oriented Development Projects
Publisher(R): SIGS Books, Inc.
Year(R): 1993
Address:
Edition:
Month:

Annotation:

From the Preface:

This book will aid technical leaders and managers responsible for building commercial software. It has two objectives: to encourage project leaders to build innovative but successful software products with higher quality and lesser risk and; to help these project leaders make interesting new mistakes on projects rather uninteresting old mistakes that have been made many times before.

Referenced for information on Lessons Learned.

Keyword: JV_Agrawal01
Reference Type: book
Author(s) -or-: Agrawal, Brij N.
Editor(s) (R):
Title(R): Design of Geosynchronous Spacecraft
Publisher(R): Prentice-Hall, Inc.
Year(R): 1986
Address: Englewood Cliffs, NJ
Edition:
Month:

Annotation:

Chapter 6 provides an overview of electrical power subsystems for geosynchronous spacecraft. It includes discussions on:

1. Solar arrays -- performance, degradation, temperature effects, and output characteristics.
2. Batteries -- # of cycles required/encountered per year at synchronous altitude, NiCd charge/discharge performance characteristics, temperature and DOD effects on life, reconditioning, as well as several graphs (such as temperature vs capacity). Also provides some info on NiH2 batteries as compared to NiCd batteries.
3. Power Control Electronics -- Explains why PCE are necessary, explains 3 different regulators: shunt, series, and pulse width modulated.
4. EPS subsystem design -- discusses major issues facing EPS design: bus voltage, single vs. dual bus, regulated vs. unregulated, and amount of fault protection
5. Miscellaneous -- outlines load for synchronous satellites, also provides several graphs (NiCd battery mass vs. power required (p. 371) and EPS masses (p. 373)

| | |
|-----------------|-----------------------------------|
| Keyword: | JV_Cost01 |
| Reference Type: | book |
| Author(s) -or-: | P. Nguyen, et al |
| Editor(s) (R): | |
| Title(R): | Unmanned Space Vehicle Cost Model |
| Publisher(R): | Space and Missile Systems Center |
| Year(R): | 1994 |
| Address: | El Segundo, CA |
| Edition: | Seventh |
| Month: | August |

Annotation:

Contains parametric cost estimating relationships built from a factual historical database.

The Minimum Unbiased Percentage Error Subsystem and Component Level CERs for determining space vehicle costs were used for Energy Generation and Energy Storage Subsystems.

Keyword: JV_DMSP01
Reference Type: manual
Title (R): DMSP Block 5D-2 System Analysis Report Volume 1: System Description
Author:
Organization: GE Astro-Space
Address: Princeton, NJ
Edition: Revision 3
Month: September
Year: 1989
:

Annotation:

This document provides the basic mission parameters for the DMSP Block 5D-2 satellites.

Data includes orbit and launch information.

Keyword: JV_DMSP02
Reference Type: manual
Title (R): DMSP Block 5D-2 System Analysis Report Volume 4: Attitude Determination and Control
Author:
Organization: GE Astro-Space
Address: Princeton, NJ
Edition: Revision 4
Month: August
Year: 1988
:

Annotation:

This volume provides a detailed outline of DMSP's attitude control system.

Keyword: JV_DMSP03
Reference Type: manual
Title (R): DMSP Block 5D-2 System Analysis Report Volume 7: Electric Power and Distribution
Author:
Organization: GE Astro-Space
Address: Princeton, NJ
Edition: Revision 2
Month: September
Year: 1989
:

Annotation:

This volume contains detailed data about the DMSP Power Subsystem. Includes battery data, launch energy, and power profiles for the DMSP Block 5D-2 spacecraft.

Keyword: JV_DMSP04
Reference Type: manual
Title (R): Block 5D-2 System Analysis Report Volume 8: Structure
Author:
Organization: GE Astro-Space
Address: Princeton, NJ
Edition: Revision 2
Month: September
Year: 1989
:

Annotation:

This volume contains technical drawings of the DMSP Block 5D-2 spacecraft.

Keyword: JV_Dunlop01
Reference Type: techreport
Author(R): Dunlop, James D., Rao, Gopalakrishna M., and Yi, Thomas Y.
Title(R): NASA Handbook for Nickel-Hydrogen Batteries
Institution(R): NASA Reference Publication 1314
Year(R): 1993
Editor:
Organization: NASA
Publisher:
Address:

Annotation:

A good collection of Nickel-Hydrogen Battery information.

Chapter 1: General Ni-H cell design. Energy densities and specific energies for various cells (p. 1-73). List of US manufacturers of Ni-H batteries (5 total as of 1993).

Chapter 2: LEO/GEO needs. Ni-H charge control. Temperature control required for Ni-H batteries. Max eclipse durations for synchronous satellites (p. 2-8). Lifetime and orbit as limiting factors justified (p.2-10). Intelsat reconditioning approach.

Chapter 3: Specifications for proven Ni-H batteries...radiator size, energy densities, size, mass, capacity, etc.

Chapter 4: Prototype battery specifications (similar to Chapter 3).

Chapter 5: Test data for Ni-H batteries.

Chapter 6: Storage and Handling concerns for Ni-H batteries.

Chapter 9: Safety/reliability information for Ni-H batteries.

| | |
|-----------------|----------------------|
| Keyword: | JV_Dyer01 |
| Reference Type: | misc |
| : | Lt. Bryan Dyer, USAF |
| : | MILSTAR Engineer |
| : | 4 SOPS |
| : | Falcon AFB, CO 80912 |
| : | Telephone Interview |
| : | 23 July 1997 |
| : | 1:45 pm |
| : | |

Annotation:

Telephone Interview with MILSTAR Engineer Lt Bryan Dyer.

DSN: 560-4827
23 July 1997
1:45 pm

MILSTAR has 4 Ni-H₂ batteries
10 year mission life

Reconditioning will save 1 V at mission end-of-life.

Each battery is off-line 60 hours for reconditioning.

Eclipse season requires additional monitoring of batteries ... 40 additional hours.

Prior to reconditioning, 8 hours of planning is required.

Reconditioning occurs twice per year (2 eclipse seasons per year).

Keyword: JV_EP01
Reference Type: manual
Title (R): Nickel Hydrogen System Technical Data Sheet
Author:
Organization: Eagle-Picher Industries, Inc.
Address:
Edition:
Month: August
Year: 1996
:

Annotation:

A collection of data sheets which provide the specifications for both a 90 A-hr and a 12 A-hr Nickel-Hydrogen battery.

Keyword: JV_Fox01
Reference Type: inproceedings
Author(R): Christopher L. Fox and Steven C. Marble
Title(R): Advances in Battery Design for Aerospace Applications
Book Title(R): 35th Aerospace Sciences Conference and Exhibit
Year(R): 1997
Editor:
Organization:
Publisher:
Address:

Annotation:

This paper details advances in recent chemical battery technologies.
Provides data for various Nickel Hydrogen batteries. We chose a 40 A-hr CPV from
this data to represent a mid-sized Ni-H₂ battery.

Keyword: JV_Hafen01
Reference Type: inproceedings
Author(R): Douglas Hafen
Title(R): Comparison of Nickel-Hydrogen and Nickel-Cadmium Reliability for Low-Earth-Orbit Batteries
Book Title(R): 27th Intersociety Energy Conversion Engineering Conference
Year(R): 1992
Editor:
Organization:
Publisher: Society of Automotive Engineers
Address: Warrendale, PA

Annotation:

This paper provides a cycle life prediction for a given reliability of 0.999. this prediction is done to compare Ni-Cd and Ni-H2 batteries for use in low-earth orbit. The results indicate that Ni-H2 batteries are more reliable than Ni-Cd batteries.

| | |
|-----------------|------------------------------|
| Keyword: | JV_Hessin01 |
| Reference Type: | misc |
| : | Captain Robert M. Hessin |
| : | |
| : | |
| : | 2d Space Operations Squadron |
| : | Falcon AFB, CA |
| : | Telephone Interview |
| : | 2 September 1997 |
| : | 2:00 pm |
| : | |

Annotation:

Telephone interview with Capt Robert M. Hessin, Chief of Analysis, 2nd SOPS (GPS).

DSN: 560-2114
2:00 pm

GPS eclipse seasons vary in duration and depend on the orbit plane.

A plane - 27 days
B plane - 32 days
C plane - 49 days
D plane - 46 days
E plane - 30 days
F plane - 30 days

Each satellite has a 12 hour period --> 2 orbits per day.

Two eclipse seasons per year.

Also, newer Block IIR spacecraft have Ni-H₂ batteries which do not require reconditioning. Older GPS spacecraft have Ni-Cd batteries and their batteries are reconditioned.

| | |
|-----------------|-----------------------|
| Keyword: | JV_Hwang01 |
| Reference Type: | misc |
| : | Warren Hwang |
| : | |
| : | |
| : | Aerospace Corporation |
| : | Los Angeles, CA |
| : | Telephone Interview |
| : | 1 August 1997 |
| : | 1:45 pm |
| : | |
| : | |

Annotation:

Telephone Interview with Warren Hwang.

(310)-336-6962
1 Aug 97
1:45 pm

Set up conference call on 5 Aug 97 at 1:00 pm est for 30 minutes to answer detailed questions.

Ni-Cd and Ni-H₂ lifetime model questions.

Charge efficiency questions (C/D ratio valid?)

Discharge efficiency questions (boost regulator only?)

Amount of sunlight portion allocated to recharge batteries?

- Depends on orbit (LEO/GEO)

Charge profile/power?

Keyword: JV_Hwang02
Reference Type: misc
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Warren Hwang

Aerospace Corporation
Los Angeles, CA
Telephone Interview
8 August 1997
11:00 am

Annotation:

Telephone Interview with Warren Hwang and Lawrence Thaller

(310)-336-6962
8 Aug 97
11:00 am

Secondary batteries fail within 15% of the predicted life. 3 sigma = 15%

For charge efficiency calculations,

Use product of C/D ratio and Watt-Hour efficiency (charge/discharge voltage differences)

For GEO orbits, a charge rate of C/15 is OK

For LEO orbits, use entire sunlight portion and charge efficiencies.

Discharge efficiencies can be captured in the boost regulator efficiencies

Availability of batteries (reconditioning)

Depends on orbit type and battery type:

- Ni-Cd in LEO (possible)
- Ni-H2 with high DOD's don't require reconditioning

| | |
|-------------------|------------------------------------|
| Keyword: | JV_Larson01 |
| Reference Type: | inbook |
| Author(s) -or-: | James R. Wertz |
| Editor(s) (R): | Wiley J. Larson and Jamer R. Wertz |
| Title(R): | Space Mission Analysis and Design |
| Chapter -and/or-: | Space Mission Geometry |
| Pages(R): | 93-127 |
| Publisher(R): | Microcosm, Inc. |
| Year(R): | 1992 |
| Address: | Torrance, CA |

Annotation:

Chapter 5 provides equations for the computation of orbital parameters.

| | |
|-------------------|------------------------------------|
| Keyword: | JV_Larson02 |
| Reference Type: | inbook |
| Author(s) -or-: | Robert Wong |
| Editor(s) (R): | Wiley J. Larson and James R. Wertz |
| Title(R): | Space Mission Analysis and Design |
| Chapter -and/or-: | Cost Modeling |
| Pages(R): | 715-740 |
| Publisher(R): | Microcosm, Inc |
| Year(R): | 1992 |
| Address: | Torrance, CA |

Annotation:

Provides cost estimating relationships and other costing information for space missions.
The non-recurring cost development factor outlined in Table 20-7 was used.

Keyword: JV_Miller01
Reference Type: techreport
Author(R): A. R. Millner
Title(R): Flywheel Components for Satellite Applications
Institution(R): MIT Lincoln Laboratory
Year(R): 16 May 1978
Editor:
Organization:
Publisher:
Address: Lexington, MA

Annotation:

Section I examines 2 different flywheels in various counter-rotating configurations.

These configurations include using the flywheels for energy storage only; momentum wheels only; and IPACS. The performance of the flywheels is compared to the performance of NiCd and NiH₂ batteries, as well as low-speed momentum wheels for use on a synchronous communications satellite (1kW payload).

Section II is a brief technology survey (1978 state of the art) of the following technologies:

Rotors
Magnetic Bearings
Touchdown Bearings
Motor/Generators
Shaft and Housing Designs

Section III lists a detailed analysis of the rotors.

Section IV analyzes the magnetic bearings.

Section V briefly discusses the motor/generator.

Keyword: JV_Neuenschwander01
Reference Type: misc
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Annotation:

Telephone Interview with Joe Neuenschwander

(402) 291-3745
1 Aug 97
10:15 am

DMSP charge efficiency for Ni-Cd batteries is difficult to determine.

C/D ratio for DMSP's eclipsing spacecraft is approximately 1.02.

There is a belief in the Ni-Cd battery community that Ni-Cd batteries must be in taper charge to be fully charged (ensures full recharge).

Provided via fax several graphs and other data for 2 DMSP eclipsing satellites.

Bus Current
Bus Voltage

| | |
|-----------------|---|
| Keyword: | JV_NMP_01 |
| Reference Type: | techreport |
| Author(R): | A.B. Chielewski |
| Title(R): | The New Millennium Program Power Technology |
| Institution(R): | Jet Propulsion Laboratory |
| Year(R): | 1996 |
| Editor: | |
| Organization: | IECEC |
| Publisher: | Institute of Electrical and Electronics Engineers |
| Address: | Piscataway, NJ |

Annotation:

This paper discuss the advantages of Lithium Ion batteries for future space missions.
It gives proposed specs for one Li-Ion battery.

Keyword: JV_Patton01
Reference Type: book
Author(s) -or-: Patton, Bruno
Editor(s) (R):
Title(R): Satellite Systems Principles and Technologies
Publisher(R): Van Nostrand Reinhold
Year(R): 1993
Address: New York, NY
Edition:
Month:

Annotation:

Chapter 9 -- Satellite Stabilization and Attitude Control

A very general description of reaction wheels, momentum wheels and magnetic bearings.

Chapter 10 -- Spacecraft Power Generation and Storage

A very general description of power conditioning electronics and batteries.

Keyword: JV_Rad01
Reference Type: misc
: Dr. Dan T. Radzykewycz, Jr.
: AFRL/VSDV
: FTA Manager, Energy Storage
: Kirtland AFB, NM
: E-mail correspondence
: 23-29 Sep 97
:
:

Annotation:

Dr. Radzykewycz provided cell information for Lithium Ion batteries.

Current Cell Data:

Average Cell Voltage: 3.5 V
Cell Mass: 0.8 kg
Cell Volume: 0.000394 cubic meters
Rated Capacity: 20 Ahr
Actual Capacity: 23 Ahr
Specific Energy: 118 Whr/kg
Energy Density: 240 Whr/liter

Probable Next Generation Cell Data:

Cell Mass: 1.46 kg
Cell Volume: 0.72 liters
Actual Capacity: 50 Ahr
Specific Energy: 140 Whr/kg
Energy Density: 285 Whr/kg

He stated that the Ahr round-trip efficiency is 0.98 for these cells. He also indicated that cell life is not dependent on depth of discharge, but temperature.

He stated that the goal is to achieve cell cycle lives of 1500 - 2000 charge/discharge cycles.

Dr. Radzykewycz also updated the Whr efficiencies of Ni-Cd and Ni-H2 batteries. Both have round-trip Whr efficiencies of roughly 0.85.

He recommended that Li-Ion batteries only be used in GEO missions. Their cycle life is not really applicable to LEO missions.

Finally, he indicated that applying a factor of 1.15 to cell mass and volume was a valid factor for aggregating cells into a battery (the factor accounts for housing and cabling).

Keyword: JV_Roland01
Reference Type: misc
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Annotation:

Phone Interview with Lt Andrew Roland
3 SOPS
DSCS III Procedures Officer

23 Jul 97
1045 am

He stated that 16 man-hours (2 full days for one person) of planning are required prior to each eclipse season for 24 batteries.

Additionally, 22 man-hours are expended over the 45 day eclipse season to monitor the reconditioning.

There are 8 spacecraft with 3 batteries per spacecraft.

Each battery is off line a total of 240 hours per reconditioning cycle.

The longest eclipse duration is 72 minutes.
Eclipse season lasts for 45 days.
There are 2 eclipse seasons per year.

Keyword: JV_Santo01
Reference Type: techreport
Author(R): Santo, Gill, Kotas, Paschall
Title(R): Feasibility of Flywheel Energy Storage Systems for Applications in Future Space Missions
Institution(R): NASA Contractor Report 195422
Year(R): January 1995
Editor:
Organization: Rockwell International, Rocketdyne Division
Publisher: NASA
Address: Canoga Park, CA

Annotation:

Chapter 7 address cost analysis comparisons for electromechanical flywheel batteries vs chemical batteries for the future space station. Sensitivity analysis is performed for several variables/scenarios (mission life, cost per lb., unit expected lifetime to name a few). Probably not useful for performing detailed costing estimates, but does provide some related info, particularly in the area where the advantages of flywheel batteries become cost effective when compared to chemical batteries.

Keyword: JV_SERI01
Reference Type: book
Author(s) -or-: Solar Energy Research Institute
Editor(s) (R): Kenneth Zweibel \and Paul Hersch
Title(R): Basic Photovoltaic Principles and Methods
Publisher(R): Van Nostrand Reinhold Company, Inc.
Year(R): 1984
Address: New York, New York
Edition:
Month:

Annotation:

A non-mathematical explanation of photovoltaic solar cells and systems. Chapter 11 (pp.168-185) provides a general discussion of the support equipment and devices that make the use of solar cells practical in a realistic environment. Although the book is primarily focuses on solar cell usage for terrestrial applications, the use of solar cells with a load, conversion equipment, load controllers and energy storage devices is covered. Chapter 11 includes several graphics depicting solar cells used in conjunction with flywheel and chemical battery energy storage systems. It also describes (in general terms) specific power conditioning equipment including inverters, power regulators, and voltage regulators. There is also a table (circa 1984) depicting characteristics of various energy storage techniques (efficiency, etc.).

Keyword: jw_Agrawal
Reference Type: book
Author(s) -or-: Brij N. Agrawal
Editor(s) (R):
Title(R): Design of Geosynchronous Spacecraft
Publisher(R): Prentice-Hall, Inc
Year(R): 1986
Address: Englewood Cliffs, NJ 07632
Edition:
Month:

Annotation:

Overview: This book discusses all aspects of the design of geo spacecraft. Of interest to us are the sections on Spacecraft Configuration, Attitude Dynamics and Control, Thermal control, and Electric Power.

Chapter1 Spacecraft Configuration: Provides a general introduction of spacecraft.

- 1.4: Spacecraft Design Considerations
- 1.6: Mass and Power Estimation

Chapter 3 Attitude Dynamics and Control: this chapter provides a overview of attitude control and methods to accomplish it.

- 3.10 Fixed Momentum Wheel with Thrusters
- 3.11 Three Axis Reaction Wheel System
- 3.13 Attitude Determination

Chapter 5 Thermal Control

- 5.2 Basics of heat transfer
- 5.3 Thermal analysis
- 5.4 Thermal control techniques
- 5.5 Spacecraft thermal design.

Chapter 6 Electric Power

- 6.3 Batteries: Discusses Ni-Cd, Ni-H₂, and Ag-Zn, charging characteristics, battery life, reconditioning.
- 6.4 Power control electronics, shunts and regulators
- 6.5 Subsystem design.

Keyword: jw Bakis
Reference Type: article
Author(R):
Title(R): Spinning Wheels for future energy - Engineers Develop Fabrication Method For Safe Flywheels
Journal(R): Engineering Penn State
Year(R): 1997
Volume: 13
Number: 3
Pages: 6-7
Month: Summer

Annotation:

NONE

| | |
|-----------------|--|
| Keyword: | jw_Gupta |
| Reference Type: | inproceedings |
| Author(R): | Gupta, Bhagwati P. and Lewis, Armand F. |
| Title(R): | Optimization of Hoop/Disk Composite Flywheel Rotor |
| Book Title(R): | 1977 Flywheel Technology Symposium Proceedings |
| Year(R): | 1978 |
| Editor: | Chang, G. C. and Stone R. G. |
| Organization: | Department of Energy |
| Publisher: | Department of Commerce |
| Address: | Washington D.C. |

Annotation:

This article presents the closed form solutions for the stresses and displacements related to a unidirectional composite rotor design.

Keyword: JW_HAVENHILL
Reference Type: misc
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August
1997

Annotation:

From SatCon

Rotor Inertia = 1.46 kg-m^2
Upper Speed = 40,000 rpm
Lower Speed = 10,000 rpm
Magnetic Bearing Average Power = 15 W
Flywheel Controller Average Power = 35 W
Motor Efficiency and Generator Efficiency
Motor Efficiency and Generator Efficiency are functions of speed and power.
I've attached a viewgraph from our PDR presentation which shows the power losses for our flywheel delivering (generator) 5.333 kW (upper curve) and being charged at 2.7 kW (lower curve). We have a MatLab/Simulink power loss model. I can send it to you if you have any use for it. In general, for a given power level the generator efficiency and motor efficiency are the same; however, the losses in the electronics are different in charge or discharge.

Reliability of one flywheel is 0.94

With regard to the weight. Our system is required to spin at launch; therefore, we did not design a separate pressure vessel. In our design the pressure vessel is integral with the structure that holds the motor stator, the auxiliary bearings, and the magnetic bearing stators. The outer shell is very light weight. Here's a weight breakdown which may help; however, you cannot just throw away the housing since it holds the other components.

Rotor 93.2 lbs
Mag Bearings 25.6 lbs
Thrust Bearing 3.4 lbs
Motor/Generator 11.3 lbs
Housing 28.8 lbs
Hardware/misc 5 lbs

| | |
|-----------------|---|
| Keyword: | jw_Hickey |
| Reference Type: | inproceedings |
| Author(R): | Hickey, J.S. |
| Title(R): | Influence of Bearing Stiffness on Flywheel-Rotor System |
| Book Title(R): | 1977 Flywheel Technology Symposium Proceedings |
| Year(R): | 1978 |
| Editor: | Chang, G. C. and Stone R. G. |
| Organization: | Department of Energy |
| Publisher: | Department of Commerce |
| Address: | Washington D.C. |

Annotation:

The bearing stiffness is important in determining the shaft critical speeds

Keyword: jw_Hoyland
Reference Type: book
Author(s) -or-: Arnljot Hoyland and Marvin Rausand
Editor(s) (R):
Title(R): System Reliability Theory Models and Statistical Methods
Publisher(R): Wiley-Interscience
Year(R): 1994
Address: 605 Third Avenue, New York, NY 10158-0012
Edition:
Month:

Annotation:

Reliability textbook for Maj Pohl's class

Keyword: jw_kirk
Reference Type: unpublished
Author(R): James A. Kirk, J.R. Schmidt, Jr., G.E. Sullivan
Title(R): An Open Core Rotator Design Methodology
Month:
Year: 1997
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Annotation:

This paper discusses the procedure followed by FARE, inc. in designing a 50 Wh open core rotator for use in a flywheel. The discussion includes how the resultant dimensions and component topology were obtained.

Keyword: jw_Leemis
Reference Type: book
Author(s) -or-: Lawrence M. Leemis
Editor(s) (R):
Title(R): Reliability, Probolistic Models and Statistical Methods
Publisher(R): Pentice Hall
Year(R): 1995
Address: Englewood Cliffs, N.J. 07632
Edition:
Month:

Annotation:

Reliability textbook from Maj. Pohl's class

Keyword: jw_Oplinger
Reference Type: inproceedings
Author(R): Oplinger D.W., and Sleptz J.M.
Title(R): Failure Characteristics of Composite Flywheels
Book Title(R): 1977 Flywheel Technology Symposium Proceedings
Year(R): 1978
Editor: Chang, G. C. and Stone R. G.
Organization: Department of Energy
Publisher: Department of Commerce
Address: Washington D.C.

Annotation:

This paper dicusses the various failure modes that can be expect to play a role in limiting the performance of composite flywheels. Also containment requirements are considered.},

Keyword: jw_Poubeau
Reference Type: inproceedings
Author(R): Poubeau, P.C.
Title(R): High Speed Flywheels Operating on "One Axis" Magnetic
Book Title(R): 1977 Flywheel Technology Symposium Proceedings
Year(R): 1978
Editor: Chang, G. C. and Stone R. G.
Organization: Department of Energy
Publisher: Department of Commerce
Address: Washington D.C.

Annotation:

The particular characteristics of magnetic suspension and of rotor designs

Keyword: jw_Riles
Reference Type: article
Author(R): Douglas M. Riles, James A. Kirk
Title(R): Design and Manufacturing for a Composite Multi-Ring Flywheel
Journal(R):
Year(R):
Volume:
Number:
Pages:
Month:

Annotation:

This paper outlines a manufacturing analysis of a composite flywheel that can be manufactured, assembled, balanced, and integrated with a magnetically suspended system test apparatus, for use in a flywheel.

The article takes about fatigue in composites

Keyword: jw_ross
Reference Type: book
Author(s) -or-: Sheldon M. Ross
Editor(s) (R):
Title(R): Introduction to Probability Models
Publisher(R): Academic Press, Inc.
Year(R): 1993
Address: 1250 Sixth Avenue, San Diego, CA 92101-4311
Edition: Fifth
Month:

Annotation:

Oper 540 book an probability models, markov chains, birth and death process, reliability, etc.

Keyword: jw_SatCon1
Reference Type: unpublished
Author(R): SatCon
Title(R): As Assesment of Flywheel Energy Storage System Reliability
Month: February
Year: 1997

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Annotation:

This report explains SatCon's work on the initial reliability assesment of a single, energy flywheel device as applicable to spacecraft power and attitude control. The outputs of the project include a reliability model, a reliability projection for a current energy flywheel configuration, suggestions for potential design modifications, and recommendations for further work.

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|-----------------|--|
| Keyword: | jw_Schlieben |
| Reference Type: | inproceedings |
| Author(R): | Schlieben, E.W., Scott R.D., Michaelis, T.D |
| Title(R): | Design Definition of a Mechanical Capacitor |
| Book Title(R): | 1977 Flywheel Technology Symposium Proceedings |
| Year(R): | 1978 |
| Editor: | Chang, G. C. and Stone R. G. |
| Organization: | Department of Energy |
| Publisher: | Department of Commerce |
| Address: | Washington D.C. |

Annotation:

This report was written under the auspices of NASA Goddard. The paper contains the results and some details of a design study and supporting analysis. The design consists of the flywheel, electrical supply, integral motor/generator and a separate power conditioning system

Keyword: jw_shaughnessy
Reference Type: techreport
Author(R): John D. Shaughnessy
Title(R): A System for Spacecraft Attitude Control and Energy Storage
Institution(R): NASA Technical Reports
Year(R): 1974
Editor:
Organization: NASA Langley Research Center
Publisher: NASA
Address: Washington, D.C.

Annotation:

The report presents a conceptual design for a double-gimbal reaction wheel/energy-wheel device which has three-axis attitude control and electrical energy storage capability.

A mathematical model for the three axis gryoscope (TAG) is developed, and a system of multiple units is proposed for attitude control and energy storage for a class of spacecraft. Control laws are derived to provide the required attitude-control torques and energy transfer while minimizing functions of TAG gimbal angles, gimbal rates, reaction-wheel speeds, and energy-wheel speed differences. A control law is also presented for a magnetic torquer desaturation system.

p.13 TAG System Concept

The TAG design consists of four rigid elements, a energy storage wheel and a seperate reaction wheel. These are inturn mounted in a gimble with two dof.

p.15 TAG and Spacecraft Control Laws

p.26 System Evaluation

p.36 Torque Equation for the TAG Unit

p.48 Gravity Gradient Torque Equation

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| Keyword: | jw_Toland |
| Reference Type: | inproceedings |
| Author(R): | Toland, Richard H. |
| Title(R): | Rotor Design Implications for Composite Material Properties |
| Book Title(R): | 1977 Flywheel Technology Symposium Proceedings |
| Year(R): | 1978 |
| Editor: | Chang, G. C. and Stone R. G. |
| Organization: | Department of Energy |
| Publisher: | Department of Commerce |
| Address: | Washington D.C. |

Annotation:

The role and needs of materials research and characterization are defined within the context of the rotor design and analysis process

Keyword: jw_Vagins
Reference Type: inproceedings
Author(R): Vagins, Milton
Title(R): Design Synthesis as Applied to Composite Flywheels
Book Title(R): 1977 Flywheel Technology Symposium Proceedings
Year(R): 1978
Editor: Chang, G. C. and Stone R. G. Chang, G. C. and Stone R. G.
Organization: Department of Energy
Publisher: Department of Commerce
Address: Washington D.C.

Annotation:

Design synthesis is defined as the achievement of a desired design criteria (constant circumferential stress), in a flywheel by preselecting a stress pattern in the loaded flywheel and then determining the variation of elastic moduli that is required to permit the desired effects

Keyword: pc_christopher
Reference Type: misc
:
: Christopher, David A.
: Aerospace Flywheel Development Program
: <http://godzilla.lerc.nasa.gov/ppo/flywheel>
: Aug
: 1997
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Annotation:

Main page

Current Status as of July 1997

-ISS program - flywheel experiment designated for development as a flight experiment.

Serves to demonstrate flywheels as a potential replacement to NiH2 batteries.

US Flywheel Systems

-Producing two flywheels for test and characterization

Prototype IPACS

-SATCON developing for LEO satellites

NAECON

-Flywheel technology attracts a great deal of interest. A technical session devoted to flywheel technology

whatis.html

Advantages of flywheels over batteries

-increase in specific energy

-longer life

-increased efficiency

-smaller volume

-power peaking capabilities

Components

Storing energy more efficiently

Usable specific energy comparison

Improvements in pointing control

-longer life

-large control torques

-large momentum storage capability

-magnetic bearing suspension reduces vibration

ipacs.html

Prototype IPACS

Composite polar weave rim + hub

>30 Whr/kg

3 wheels, CMG-type configuration

Status - Phillips Lab Phase 2 STTR to SATCON

To be completed end of FY98

program.html

Challenges of developing a flywheel for IPACS

-Safety

-Rotor Design

-Magnetic Bearings

Potential Applications

-LEO Satellites

-GEO Satellites

-Space Station

-Planetary probes

- Aircraft

-Military vehicles

-Hybrid and electric vehicles

-Uninterruptable Power Supplies (eg., KSC launch operations)

-Power peaking (load leveling)

Keyword: pc_flatley
Reference Type: article
Author(R): Thomas W. Flatley
Title(R): Tetrahedron Array of Reaction Wheels for Attitude Control and Energy Storage
Journal(R): Society of Automotive Engineers
Year(R): 1985
Volume:
Number:
Pages:
Month:

Annotation:

Describes using 4 flyhweel batteries to provide attitude control and energy storage. The system employs a tetrahedral configuration. In sunlight, the system provides voltage regulation as the wheel are accelerated to the maximum speed. In the dark, the flywheels provide regulated power to the spacecraft electrical loads. A microprocessor controls the field current levels for the four flywheels based on system voltage level, flywheel speeds, and desired attitude control torques.

Keyword: pc_ford
Reference Type: unpublished
Author(R): Kevin A. Ford
Title(R): Gimbaled Momentum Wheel via Angular Momentum Methods
Month: Nov
Year: 1996

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Annotation:

A development of Gimbaled momentum wheel kinematics.
Section 2.6 summarizes the final equations of motion. Section 2.7 looks at the specific case of gimbal angles constant, or a momentum wheel case. Section 2.8 looks at the constant wheel speed case, typically referred to as Control Moment Gyros.

Keyword: pc_hall1
Reference Type: unpublished
Author(R): Chris D. Hall
Title(R): High Speed Flywheels for Integrated Energy Storage and Attitude Control
Month: April
Year: 1997

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Annotation:

Uses singular value decomposition to show that with 4 or more flywheels energy storage and attitude control can be decoupled.

Keyword: pc_honeywell
Reference Type: booklet
Title(R): Space Systems Momentum Control
Author: Honeywell
How Published: Promotional material
Address: Space Systems, Honeywell Inc., Phoenix, Arizona
Month: Aug
Year: 95
:
:

Annotation:

Promotional material for Honeywell's momentum and reaction wheel and control moment gyro assemblies.

Performance characteristics for CMGs include:

Angular momentum, rotor speed, spin motor torque, output torque, power, dimensions and mass

Performance characteristics for reaction and momentum wheels include:

Angular momentum, output torque, peak and steady state power, wheel speed, weight, dimensions, and life.

Keyword: pc_hughes
Reference Type: book
Author(s) -or-: Peter C. Hughes
Editor(s) (R):
Title(R): Spacecraft Attitude Dynamics
Publisher(R): John Wiley & Sons
Year(R): 1986
Address:
Edition:
Month:

Annotation:

Discusses attitude dynamics for spacecraft and includes discussions on gyrostats (spacecraft with wheels) and stability concerns

Ch 2: Rotational Kinematics

2.1 Covers reference frames and rotation matrices

2.2 Angular displacement parameters, covering Euler parameters and angles, and infinitesimal displacements

Table 2.1 Euler Angle Rotation Matrices

2.3 Equations for angular velocity, for direction cosines, axis/angle parameters, Euler angles and parameters, infinitesimal displacements

Table 2.3 Comparison of Parameter Alternatives

Ch 3: Attitude Motion Equations

Equations for point mass, system of point masses, rigid body, rigid body with damper, rigid body with wheel.

pg 69. Rigid body with wheel equations

Ch 4: Attitude Dynamics of a Rigid Body

Provides equations for torque free motion; axisymmetric and tri-inertial. Tri-inertial case is in the form of elliptic functions.

Stability for a rigid body looks at Liapunov's method for omega stability and infinitesimal analysis for attitude stability

Ch 5: Effect of Internal Energy Dissipation on Directional Stability of Spinning Bodies

Ch 6: Directional Stability of Multispin Vehicles

Very useful section deals with spinning wheels and the effects on directional stability.

Three cases are reviewed: zero momentum in the rigid body, zero momentum in the total system, and the general case. Equations of motion are provided for all three cases.

Ch 7: Effect of Internal Energy Dissipation on Directional Stability of Gyrostats

Two methods are examined: energy sink analysis and discrete dampers

Ch 8: Spacecraft Torques

Covers 4 sources of environmental torques: gravitational, aerodynamic, solar radiation, and magnetic. Other sources of torques are also discussed.

Ch 9: Gravitational Stabilization

Effects of gravity on spacecraft in detail. Lagrange region, DeBra-Delp, pitch motion, effects of elliptic orbits, and damping effects are all discussed. Flight experience is provided.

Ch 10: Spin Stabilization in Orbit

Equations for examining stabilization. Also discussed are external torques averaged over one orbit. Flight data and long term effects are also discussed.

Ch 11: Dual Stabilization in Orbit

For gyrostats, dynamical equilibria and Roberson relative equilibria are covered.

Librations about equilibria are also discussed. Stability of cylindrical and noncylindrical equilibrium are also provided. Other topics on gyrostats are discussed briefly:

wobble dampers, bearing alignment, attitude acquisition and reacquisition, nonlinear phenomena and active control. Bias momentum satellites and examples are discussed.

Appendix A: Elements of Stability Theory

Stability of the origin and linear approximation theories are provided. Liapunov's

method is discussed and theorems for checking stability are provided.

| | |
|-----------------|---|
| Keyword: | pc_ithaco |
| Reference Type: | misc |
| : | Cohen, J. |
| : | Ithaco home page |
| : | http://www.newspace.com/industry/ithaco |
| : | Nov |
| : | 1996 |
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| : | |

Annotation:

Description of Ithaco products

E-wheel.html

Ithaco's E-wheel reaction and momentum wheel assemblies

Flight Schedule

-12 E-wheel type reaction or momentum wheels have been delivered or are in various stages of manufacture.

E-wheel Performance characteristics

E-wheel Performance Characteristics
Includes momentum storage, torque capability, mass, power, life, volume.

T-wheel.html

T-wheel is a momentum/bidirectional reaction wheel.

Flight Schedule

-86 Type A and Type B T-wheels have been delivered or are in various stages of manufacture.

T-wheel Performance characteristics

Includes momentum storage, torque capability, mass, power, reliability, volume.

Keyword: pc_kaplan
Reference Type: book
Author(s) -or-: Marshall H. Kaplan
Editor(s) (R):
Title(R): Modern Spacecraft Dynamics and Control
Publisher(R): John Wiley & Sons
Year(R): 1976
Address:
Edition:
Month:

Annotation:

Ch 1: Introduction

Ch 2: Fundamental Spacecraft Dynamics

Ch 3: Orbital Maneuvers

Ch 4: Attitude Maneuvers

4.1 Momentum Precession

4.2 Reorientation with Constant Momentum

4.3 Attitude Determination

Ch 5: Attitude Control Devices

5.1 Gyroscopic instruments

5.2 Momentum exchange techniques

5.4 Magnetic torquers

5.5 Gravity gradient stabilization

Ch 6: Automatic Attitude Control

6.1 Linear control theory

6.2 Design of a bias momentum system

6.3 Design of an all-thruster system

Ch 7: Fundamentals and Methods of Astrodynamics

Ch 8: Orbital Perturbations

Ch 9: Special Problems

9.1 Attitude acquisition maneuver of a bias momentum satellite

9.2 Automatic detumbling of a space station

9.3 Yaw sensing strategy for inclination control

Keyword: pc_shuster
Reference Type: article
Author(R): Malcolm D. Shuster
Title(R): A Survey of Attitude Representations
Journal(R): The Journal of Astronautical Sciences
Year(R): 1993
Volume: 41
Number: 4
Pages: 439-517
Month: October-December

Annotation:

A thorough application of various attitude representations.
Includes 400+ equations and 163 references.
Covers orthogonal transformations, rotation matrices, Euler angles, axis-azimuth representation, Euler-Rodrigues parameters and quaternions, Rodrigues parameters, Cayley-Klein parameters, and modified Rodrigues parameters.
Attitude kinematics and errors are covered.
Properties of quaternion transformations are also covered.

Keyword: pc_wiesel
Reference Type: book
Author(s) -or-: William E. Wiesel
Editor(s) (R):
Title(R): Spaceflight Dynamics
Publisher(R): McGraw-Hill Book Company
Year(R): 1989
Address:
Edition: 1st
Month:

Annotation:

Ch 1: Particle Dynamics

Ch 2: Two-Body Problem

Ch 3: Earth Satellite Operations

Ch 4: Rigid Body Dynamics

Ch 5: Satellite Attitude Dynamics

Ch 6: Gyroscopic Instruments

Ch 9: Space Environment

Keyword: sf_agrawal
Reference Type: book
Author(s) -or-: Agrawal, Brij N.
Editor(s) (R):
Title(R): Design of Geosynchronous Spacecraft
Publisher(R): Prentice-Hall Inc.
Year(R): 1986
Address: Englewood Cliffs, NJ
Edition:
Month:

Annotation:

336: The optical (noncell) losses, which consist mostly of coverglass adhesive darkening, are relatively large, 4 to 10%, during the first year in orbit, and increase very little thereafter (included in Design and Assembly losses?)

The solar cell damage is about 3% during the first year and approximately 15% for 7 years of operation. (similar numbers in other sources)

Approximately one-half of the total degradation occurs in the first two years of a 7-year orbit life.

Extending life to 10 years results in an additional 5% loss only (3% annual rate).

339: Measured data show that the solar cell maximum available power will fall off faster than the cosine of the angle of incidence indicates. The principal factor that causes this deviation is the change in reflection coefficients at large angles.

342: GaAs cells of 17% efficiency have been produced, compared to 15% efficiency for the best available silicon solar cells (similar numbers).

A typical Si cell degrades approx. 0.5% for every degree C rise in temp. GaAs's degradation is approx. half that (agrees with SMAD).

At EOL, K7 Si cells have 14 mW/cm² and 7mW/cm² for 28 deg C and 128 deg C, a 50% degradation due to temp.

For GaAs, 19.08 mW/cm² and 13.8 mW/cm², a mere 27% degradation.

Hence, GaAs cells are more attractive at higher temps, such as in a solar concentrator system.

As a result of annealing, the EOL efficiency of GaAs may be very close to that of BOL efficiency (not supported elsewhere).

The GaAs cell mass for a similar-size cell is approximately twice that of a Si cell.

Ex. BOL 1kW GaAs array
27% less area and 7% less mass than the most efficiency Si array.
(Using temp degradation, numbers work.)

Individual cells are arranged in series to provide the desired voltage and in parallel to provide the desired current requirements.

346: Solar booms ranging between 2.2 cm and 3.4 cm in diameter.

347: Solar intensity at aphelion = 1309 W/m², 4 July

Inclination of equator at solstices = 23.44 deg

Worst case, combining inclination of equator with distance from Sun
= summer solstice, 1311 W/m²
incl. 23.44 deg
power=0.8885 x 1353 = 1202 W/m²

July 4 is just as poor
1309 W/m² incl.=23.13 deg
power=0.889 x 1353=1203 W/m²

The design requirement for a solar array is generally to produce the current and voltage demanded at the EOL summer solstice condition.

| | |
|-----------------|------------------------------|
| Keyword: | sf_allen1 |
| Reference Type: | misc |
| : | Slides from Douglas M. Allen |
| : | W.J. Shafer Associates, Inc. |
| : | Cleveland, OH |
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Annotation:

SCARLET = solar concentrator arrays with refractive linear element technology
23% efficiency
recurring cost < \$500/W vs. \$1000/W for Si
specific power > 50 W/kg vs. 25 W/kg for GaAs
prime power on New Millennium Deep Space mission

Case example using SBIRS Low
1.65 kW satellite
1,500 km orbit
7 year life
500 kg satellite

Keyword: sf_bennet1
Reference Type: article
Author(R): Bennet, Gary L.
Title(R): Electrical Power for Spacecraft: Opinions and Issues Part 1: Solar Power and Energy Storage
Journal(R): Space Times
Year(R): 1997
Volume:
Number:
Pages:
Month: May-June

Annotation:

Energy generation and energy storage
Specific energies, densities, etc.
Alternatives

solar energy
fuel cells
batteries
flywheels

current performance
emerging technology
capabilities
limitations
issues
solar array savings

9: Depending upon the mission, the electrical power subsystem (EPS) can take up as much as 25 percent of the mass of a spacecraft.

More power from the same EPS mass can be used to drive electric thrusters which offer better performance. Thus electrical power is synergistic with on-board propulsion.

10: areal power density

11: gallium arsenide becoming the technology of choice

Pointing concentrator cells toward the Sun requires more attention to attitude control than normal solar arrays.

12: The choice of solar array technology depends upon a number of factors including the power level, stowage volume, and environmental conditions.

15: Recent improvements in the electrolyte have allowed NiH2 batteries to be considered for LEO applications with greater depths of discharge and longer lifetimes.

16: GRAPH: Solar Flux as a Function of Distance from the Sun

Flywheel advantages:

- they are smaller in volume than NiH2 batteries
- shorter manufacturing lead times
- no taper charging
- no reconditioning
- precise measurement of SOC

Keyword: sf_boeing1
 Reference Type: techreport
 Author(R): Boeing Aerospace Co.
 Title(R): Flywheel Technology Development Program for Aerospace Applications
 Institution(R):
 Year(R): 1997
 Editor:
 Organization: Boeing
 Publisher: NASA Lewis Research Center
 Address:

Annotation:

- 5: advantages and disadvantages
- 9: space stations requirements
- 11: importance of energy storage efficiency, more benefits of flywheels
- 15: efficiency of flywheel ESS
- 21: utilization of excess sunrise power, a benefit
- c4: flywheel rotors, weights
- 23: specific strength
- 27: flywheel performance predictions, energy densities
- 41: flywheel rotor containment
- 41: deferring spin-up until after launch
- 44: flywheel out-of-balance considerations
- 44: wheels operating at low max. stress
- c5: motor/generators, combined vs. separate, weights
- 51: permanent magnetic materials, electromagnetics
- 52: temperature control considerations
- c6: bearings
- 63: mechanical bearings, may be appropriate for gimbaled wheels
- 63: bearing life
- 66: magnetic bearings
- 72: comparison of mechanical and magnetic bearings
- c7: FESS vs. batteries
- 74: weight assessment, DoD
- 78: speed range
- 78: Thermal control
 FES has higher efficiency, less heat to dissipate, higher temps, heat load more uniform with time
 Batteries have greater transient heat storage capability
- 78: PMAD
 FES does not need a separate charge controller
 voltage regulation of the FES is very fine and essentially provides a regulated bus – higher power efficiency, lower weight, smaller solar array size
- 85: and less fuel to counteract drag

74: 100% DoD, design DoD, emergency capability

80: reliability, derating factors

86: peak power with FES, solar arrays can be sized closer to average power and can be supplemented often by FES

88: safety -- any rotor that can burst is considered an unsafe design

89: FES has a poor reserve capability

Keyword: sf_christopher1
Reference Type: unpublished
Author(R): Christopher, David A. and Beach, Raymond
Title(R): Flywheel Technology Development Program for Aerospace Applications
Month: February 1
Year: 1997

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Annotation:

- 2: history of flywheel research
- 3: flywheel applications and benefits; peak power requirements; specific energy at battery, energy storage system, and spacecraft level
- 4: IPACS
- 5: graphs of specific energy
- 7: deep depth of discharge benefits
- 10: load leveling
- 11: component technologies
- 12: remaining challenges
- 13: FASTPAC -- design and fabricate energy storage system
- 13: SatCon -- design and fabricate IPACS
- 14: TRW and U.S. Flywheel Systems, Inc. -- FES system engineering model
- 15: NASA LeRC and Boeing -- flight demonstration
- 16: 2d flight demonstration
- 17: bearing systems
- 18: materials and structures
- 18: power electronics
- 18: system integration
- 19: safety and reliability, DARPA, flywheel rotor burst events

Keyword: sf_larson1
 Reference Type: book
 Author(s) -or-: Larson, Wiley J. and Wertz, James R.
 Editor(s) (R): Wertz, James R.
 Title(R): Space Mission Analysis and Design
 Publisher(R): Microcosm, Inc.
 Year(R): 1992
 Address: Torrance, CA
 Edition: Second
 Month:

Annotation:

49: Mission utility analysis, the process of quantifying how well the system meets its overall mission objectives.

66: The purpose of the trade studies and utility analysis is to make the decisions as informed as possible.

We wish to add quantitative information to the decisions, not quantify the decision making. We should not undervalue the decision maker's judgement by attempting to replace it with a simplistic formula or rule.

101: Analyzing eclipses for a LEO.

104: Most of the time, the eclipse duration will be close to the maximum.

108: Equation for Earth coverage angle, needed to compute max. eclipse length.

287: Description of an ACS and functions.

288: Description of a power system.

296: Body mounted vs. panel mounted arrays.

Solar arrays produce about 100 W/m^2 (projected area) unregulated power, given 7% efficiency.

297: Ratio of total array area to projected area:

- pointed towards Sun = 1
- cylindrical arrays with axis perpendicular to Sun

or degradation in the solar array. This degradation depends on orbit altitude and radiation environment. 30% is typical for 10 years at GEO. We can assume the same value for altitudes of 800 km or less. Between these altitudes the degradation is much larger.

303: ADCS description.

Possible design requirements:

- pointing accuracy, pointing stability, slew rate.

Spin stabilization vs. 3-axis control.

Possible sensors.

Flywheel types.

306: Systems which use wheels can avoid dead-zone attitude errors, so they usually can operate close to the sensor accuracy.

306: Torquing methods.

Variable-speed reaction wheels produce only limited control torque (less than 1 Nm). CMGs can develop torques up to several thousand Nm.

307: The torquing ability of (an ACS) may be set by an acceleration requirement such as that arising from an attitude slew maneuver...

Equations...

Angular impulse capability and requirements.

308: Disturbance torques

311: Weight and Power of ADCS Components.
Equations...

316: Power subsystem functions

317: Solar arrays.

Temperature effects favor cylindrical arrays, so the actual ratio of total area to projected area is 2.5.

Solar constant is 1358 W/m^2 .

Area of array = $P/(\text{efficiency} \times \text{solar constant})$

Mass of array = $P/\text{specific performance}$

Current designs have specific performances between 14 and 47 W/kg at EOL.

The high end would provide 66 W/kg at BOL.

Solar arrays mounted on the spacecraft's body usually weigh less than planar arrays.

318: Batteries:

- have complex wear-out mechanisms, limiting cycle life as a function of depth-of-discharge.
- Temp, charge rate, discharge rate, and over-charging affect cycle life.
- How to determine capacity.

Specific energies for NiCd and NiH₂.

By using several small batteries, redundant batteries can be added with less weight penalty than for a second large battery.

Regulation.

PMAD mass, efficiency.

324: FMECA techniques.

340: Purpose of ADCS.

Disturbance torques.

Gyroscopic stiffness of spinning body.

341: Slew maneuvers.

External references.

342: ACS design.

- 1a. Define control modes
- 1b. Define requirements
2. Select control type
3. Quantify disturbances
4. Select ADCS hardware
5. Define ADC algorithms

344: Pointing requirement examples.

345: Selection of control type

- passive control
 1. gravity gradient
- orientation around nadir is unconstrained
- momentum wheel in pitch axis to control yaw axis???
- 2. energy dissipation for stability
- 3. magnets

346: AC methods and capabilities.
spin stabilization.

347: Dual-spin.

Precession.

Wobble.

Nutation.

348: Nutation dampers.
3-Axis control.
Zero-momentum.
Momentum dumping.
Momentum bias.

349: Roll-yaw coupling.

Keyword: sf_naecon97
Reference Type: misc
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Annotation:

Tom Pieronek, TRW:

- People looking at 15-year technologies for cost effectiveness.
- Batteries are life-limiting factor at 7 years.
- Flywheels save 200W from a 3,650W solar array.
 - 80% round-trip efficiency vs. 75% for batteries
 - no taper charging
- As one system reaps benefits, the other has to compensate.

Doug Havenhill, SatCon:

- Energy storage and power capability are decoupled with flywheels. Currently, you must increase battery size for either.
- Voltage regulation and charge/discharge control are built into the power control electronics. Have to have PCE anyway.

John Edwards, Boeing:

- Contingency ES sizing for anomolous discharging

- Same alternatives we are considering
- Counter-rotating wheels maintain acceptable torque limits
- Any failure that produces a large torque (e.g. wheel breaking) must be negated

Life Issues (TRW)

- Cycle life has not been demonstrated for space applications
 - Protoflight designs initiated
 - Reliability analysis/FMEA initiated
- DARPA life test program initiated
 - life cycle tests
 - failure/destruction tests

ACS Issues (TRW)

- Momentum due to rotor inertia mismatch
 - Use differential wheel speeds to cancel
- Rotor inertia variations over life of flywheel
 - creep, fatigue
 - rotor/bearing balance shifts

Configuration Ideas (TRW)

- Four pairs of counter-rotating wheels in pyramid configuration
- Energy stored in 6 of 8 wheels
- Any single wheel causes shutdown of its counterpart

PMAD, Dr. Reinhardt, PL

Averaging data from DSCS III, DSP, GPS, and, of course, MILSTAR:

- The Electrical Power System (EPS) comprises 20-30% of the satellite's mass.
- The EPS mass is broken down as follows:
 - Solar array and structure 30%
 - Energy storage 25%
 - PMAD cabling and harness 25-35%
 - PMAD electronics 10-20%
- Mass equations in PMAD slides include: solar arrays, electronics, thermal control, harness

Keyword: bj Dinwoodie
Reference Type: techreport
Author(R): Dinwoodie, Thomas L.
Title(R): Flywheel Storage for Photovoltaics: An Economic Evaluation of Two Applications
Institution(R): MIT Energy Lab Report no. MIT-EL-80-002
Year(R): 1980
Editor:
Organization: MIT/Lincoln Labs
Publisher:
Address:

Annotation:

Chap 1 - INTRO: The people at MIT and Lincoln Labs got together and analyzed the use of flywheels with photovoltaics(PV) to support a single family residence and a multi-family load center. A large portion of this paper works with homes and dwelling related energy concerns. Yet, there appears to be some useful data for our project, in terms of basic flywheel information and assumptions.
9: Figure of basic systems flow structure for flywheels and batteries.

Chap 2 - Study Org: Gives assumptions and the basic data for the underlying calculations in the study.
13: Basic flywheel and PV efficiency assumptions.
19: Flywheel and PV cost assumptions.
Discussion of IPACS concept and a rough hack at the savings.
20: Flywheel overall capacity and efficiency. Estimate of .62 efficiency.

Chap 3 - Results: Not very much pertinent data in this chapter. Deals more with local electrical costs and basic knowledge of the systems.
74-79: Discussion about the sensitivity of the results to flywheel efficiency.

Chap 4 - Discussion: Analyzed in terms of a goal to increase energy storage, not meet a requirement better. Therefore, much of this discussion is not relevant.

Chap 5 - Conclusions: Not significantly pertinent to our project.

| | |
|-----------------|--------------------------------------|
| Keyword: | bj Eschbach |
| Reference Type: | book |
| Author(s) -or-: | |
| Editor(s) (R): | Eshbach, Ovid W. |
| Title(R): | Handbook of Engineering Fundamentals |
| Publisher(R): | John Wiley & Sons |
| Year(R): | 1975 |
| Address: | |
| Edition: | Third |
| Month: | |

Annotation:

Great reference for all types of generic and specific engineering theories and equations.
Top notch resource for all source info.

| | |
|-----------------|---------------------------|
| Keyword: | bj Jennings |
| Reference Type: | book |
| Author(s) -or-: | |
| Editor(s) (R): | Jennings, Roger |
| Title(R): | Using Microsoft Access 97 |
| Publisher(R): | Que |
| Year(R): | 1997 |
| Address: | |
| Edition: | Special |
| Month: | |

Annotation:

Detailed guide for Access 97 applications. Good tool details for the details in using Access with Visual Basic. Great reference.

Keyword: bj Keckler
Reference Type: techreport
Author(R): Keckler, Claude R.
Title(R): Integrated Flywheel Technology
Institution(R): NASA Conference Publication 2290
Year(R): 1983
Editor: Keckler, C. R.
Organization: Langley Research Center
Publisher: NASA Scientific and Technical Information Office
Address: 21000 Brookpark Road, Cleveland, Ohio, 44135

Annotation:

OVERVIEW: This is the presentation given by several folks at the two-day integrated flywheel system workshop. It took place at NASA Goddard on Aug 2-3, 1983. Each number is a new presentation.

- 1) IPACS: Discussion of IPACS concept and a rough hack at the savings.
- 16-17,20: Lists of IPACS hardware, benefits, and large CMG characteristics.
- 2) GSFC Flywheel status: Discusses the effort at GSFC for assessing flywheels.
- 24: Flywheel drawing.
- 25: Potential Advantages.
- 27: Power system configuration comparison. Lists the need to lock wheels during launch.
- 29: Summary of critical technolgis to make an integrated flywheel design work. States that the system must be integrated in order for the advantages to be worth it.
- 30-32: Good charts on critical technologies.
- 3) Overview of State-of-the-Art Flywheel technology: Oak Ridge National Lab.
- 38: Energy storage density comparison table.
- 4) European Integrated Flywheel Technology:
- 5) Electrical Power Trade-Offs:
- 6) Space Station Energy Sizing:
- 53-62: Good comparisons on efficiency and weight. Efficiency equation.
- 7) Boeing Flywheel study:
- 8) Space Station Control...:
- 88: Chart of efficiency verse solar array size.
- 90: Adv. of IPACS over batteries and fuel cells.
- 9) Combined Attitude Control and Energy Storage: Configuration Options for wheels.
- 10) IPACS Attitude Control Technology Considerations:
- 102-103: DG CMG versus reaction wheel.
- 11) Flywheel Electronics:
- 114-115: Efficiency overall & charge/discharge efficiencies.
- 12) IPACS Electronics - Comments on LaRC efforts:
- 120-121: Motor and Generator operation discussed.
- 13) Angular Momentum Control Device (ACMD):
- 14) Magnetic Bearings for Inertial Energy Storage:

136: Upper limit of mag bearings at 230 psi. Discusses in more detail.

153: Summary table of storage and control options.

171-175: Good battery Info

175-189: Decay and failure info for both batteries and flywheels.

182: Three types of energy storage devices drawn and discussed.

184: Power system mass breakdown chart. Lists several options.

Keyword: bj Keckler2
Reference Type: techreport
Author(R): Keckler, Claude R
Title(R): Description and Simulation of an Integrated Power and Attitude Control System Concept for Space-Vehicle Application D-7459
Institution(R): NASA Technical Notes; D-7451 - D-7460
Year(R): 1974
Editor:
Organization: Langley Research Center
Publisher: NASA Scientific and Technical Information Office
Address: Hampton, Va. 23665

Annotation:

SUMMARY: An IPACS system is discussed. A shuttle launched Research and Applications Modules (RAM) A303B Solar Observatory is used as the pointing requirements driver. A simulation is performed and it is found that the IPACS concept can meet the RAM pointing requirements even in a failure-mode configuration.

INTRO: Basic ACS and Energy storage discussion.
2: CMG for SKYLAB

SYSTEM DESCRIPTION: More relevant basic flywheel discussion. Uses two rotors to simulate a three rotor system with one failed.
6-7: Schematic diagrams of IPACS

EXAMPLE MISSION CHARACTERISTICS: Discussion of meeting pointing requirements and adverse interactions produced by torques generated during power-transfer ops.

SIMULATION DESCRIPTION:
12: Functional block diagram of IPACS simulation.

HARDWARE DESCRIPTION: Describes the two double-gimbaled model CMG units used in simulation.

DISCUSSION OF RESULTS: Discusses the means of meeting pointing accuracy based on simulation data.

CONCLUDING REMARKS: Good discussion to read. May be pertinent to our efforts.

APPENDIX A: Equations, Scaling, and computer algorithm for simulation.

APPENDIX B: Fortran code for two CMG steering law.

Keyword: bj Kowal
Reference Type: techreport
Author(R): Kowal, Michael T
Title(R): Mechanical System Reliability for Long Life Space Systems
Institution(R): Comet
Year(R): 1994
Editor:
Organization: NASA Lewis Research Center
Publisher: NASA Scientific and Technical Information Office
Address: 21000 Brookpark Road, Cleveland, Ohio, 44135

Annotation:

OVERVIEW: The report provides a reference for application of first order reliability methods to space systems. Good source of listings for other literature on the topic.

Chap1: Overview of failure/limit states and environmental factors influencing failure. Good big picture for failure analysis background.

Chap2: Discussion on general formulas for types of wear, corrosion, fatigue, and thermal degradation of materials. Not great detail but many areas covered

27-50: Corrosion models may be pertinent to battery failure results.

52: High cycle fatigue information. Good starting point.

57: Thermal degradation general info.

Chap3: Discussion of first order reliability methods.

Chap4: Linking Reliability methods to Life-cycle costs. This chapter may prove very helpful.

71: Life cycle cost element chart

74: Taguchi/Design of Experiment Methods for cost variables.

78: Steps to integrating cost and reliability into design. May be very useful!

80: Flow chart for integration of reliability and costs (see 78 above).

APPENDIX1/2: Provides fortran files to perform some of the methods listed above.

APPENDIX3: Review of the research the author performed.

Keyword: bj Lamport
Reference Type: manual
Title (R): Latex User Guide and Reference Manual
Author: Lamport, Leslie
Organization: Digital Equipment Corp
Address:
Edition: Fifth Printing, Addison-Wesley
Month:
Year: 1986
:

Annotation:

A easy to use guide for writing in Latex. Provides great examples and a cheat sheet.

Keyword: bj Microsoft
Reference Type: manual
Title (R): Visual Basic, Getting Started
Author:
Organization: Microsoft Corp
Address:
Edition:
Month:
Year: 1997
:

Annotation:

Overview text for Visual Basic 5. Basic VB info.

Keyword: bj Microsoft2
Reference Type: manual
Title (R): Visual Basic, Component Tools Guide
Author:
Organization: Microsoft Corp
Address:
Edition:
Month:
Year: 1997
:

Annotation:

A detailed guide on the tools and components available in Visual Basic 5. A very good reference for understanding a functional area or tool of VB5.

Keyword: bj Microsoft3
Reference Type: manual
Title (R): Visual Basic, Programmer's Guide
Author:
Organization: Microsoft Corp
Address:
Edition:
Month:
Year: 1997
:

Annotation:

A very detailed guide on how to program with Visual Basic 5. Excellent source of info for "how to" guidance.

Keyword: bj Microsoft4
Reference Type: manual
Title (R): Visual Basic, Guide to Data Access Objects and Crystal Reports
Author:
Organization: Microsoft Corp
Address:
Edition:
Month:
Year: 1997
:

Annotation:

NONE

| | |
|-----------------|--------------------------|
| Keyword: | bj Person |
| Reference Type: | book |
| Author(s) -or-: | |
| Editor(s) (R): | Person, Ron |
| Title(R): | Using Microsoft Excel 97 |
| Publisher(R): | Que |
| Year(R): | 1997 |
| Address: | |
| Edition: | Special |
| Month: | |

Annotation:

Detailed guide for Excel 97 applications. Good tool details for the details in using Excel with Visual Basic. Great reference.

| | |
|-----------------|-----------------------|
| Keyword: | bj Pfaffenberger |
| Reference Type: | book |
| Author(s) -or-: | Pfaffenberger |
| Editor(s) (R): | |
| Title(R): | Web Search Strategies |
| Publisher(R): | MIS Press |
| Year(R): | 1996 |
| Address: | |
| Edition: | First |
| Month: | |

Annotation:

NONE

Keyword: bj Santo
Reference Type: techreport
Author(R): Santo, G. Espiritu
Title(R): Feasibility of Flywheel Energy Storage systems For Applications in Future Space Missions
Institution(R): Comet
Year(R): 1995
Editor:
Organization: Rockwell for NASA Lewis Research Center
Publisher: NASA Scientific and Technical Information Office
Address: 21000 Brookpark Road, Cleveland, Ohio, 44135

Annotation:

OVERVIEW: This is the final contractor report by Rockwell for NASA LaRC. It examines the overall feasibility of flywheels as energy storage devices. Specifically geared toward the Space Station Alpha, yet contains some good data. It incorporates info from the previous Rocketdyne flywheel studies. Several comparisons to batteries. Good source of definitions for acronyms.

Chap1: Overview of the space station flywheel concept.

6: Life cycle cost comparison between a generic battery and the flywheel.

Chap2: Generic requirements and info.

Chap3: Space station overview.

11-13: Space Station battery characteristics and requirements tables. Thermal requirements also.

13-17: General comparison tables.

20: Rough table on comparing batteries vs. flywheels in performance characteristics (energy density, efficiency, expected life)

Chap4: Covers Rocketdyne flywheel and the motor/generator options.

Chap5: Covers the means of putting the flywheel ORU into the station (not really relevant).

37: Thermal stuff and efficiency info.

38-42: Failure analysis info.

Chap6: More space station info (N/A)

Chap7: Flywheel life cycle cost analysis. Seems to be good stuff for us. Spreadsheet model. Might be able to utilize in our model in some way.

Chap8: Flywheel Development Plan (N/A)

Chap9: Conclusions

APPENDICES: Quite a bit of data and plots in the appendix. Worth scanning.

| | |
|-----------------|--------------------------|
| Keyword: | bj Urban |
| Reference Type: | book |
| Author(s) -or-: | Urban, Michael |
| Editor(s) (R): | |
| Title(R): | An Introduction to LaTeX |
| Publisher(R): | Tex Users Group |
| Year(R): | 1986 |
| Address: | |
| Edition: | |
| Month: | |

Annotation:

A short reference for LaTeX. Quick reading for a small overview of how LaTeX works.

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5. Eagle-Picher Industries, Inc. *Nickel Hydrogen System Technical Data Sheet*, August 1996.
6. Fox, Christopher L. and Steven C. Marble. "Advances in Battery Design for Aerospace Applications." *35th Aerospace Sciences Conference and Exhibit*. 1997.
7. GE Astro-Space, Princeton, NJ. *DMSP Block 5D-2 System Analysis Report Volume 7: Electric Power and Distribution* (Revision 2 Edition), September 1989.
8. Hafen, Douglas. "Comparison of Nickel-Hydrogen and Nickel-Cadmium Reliability for Low-Earth-Orbit Batteries." *27th Intersociety Energy Conversion Engineering Conference*. Warrendale, PA: Society of Automotive Engineers, 1992.
9. Hall, Arthur D. "Three-Dimensional Morphology of Systems Engineering," *IEEE Transactions on Systems Science and Cybernetics*, SSC-5(2):156-160 (April 1969).
10. Hall, Chris D. "High Speed Flywheels for Integrated Energy Storage and Attitude Control." Apr 1997.
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Vita

Captain Steven A. Fischer was born on 18 February 1968 in Hamilton, Ohio. He graduated from Ross High School there in June 1987. In the Fall of 1987, he entered The George Washington University, and graduated in May 1991 with a Bachelor of Science degree in Systems Analysis and Engineering. Upon graduation, he was commissioned as a Second Lieutenant in the United States Air Force.

After graduating from Undergraduate Space Training at Lowry Air Force Base, Colorado as Distinguished Graduate, he began his first assignment at Falcon Air Force Base, Colorado. There he attended the 50th Crew Training Training (50 CTS) and graduated as Distinguished Graduate. He then worked as a Satellite Crew Commander for the 3rd Space Operations Squadron (3 SOPS). He was responsible for the command and control of Defense Satellite Communications System, Model III (DSCS III) and Military Strategic, Tactical, and Relay System (MILSTAR) satellites. He also served as Chief, MILSTAR Training; Executive Officer; and Chief, Tactics, Plans, and Requirements. In May 1996, he entered the Graduate Space Operations program in the School of Engineering at the Air Force Institute of Technology.

His next assignment is to the National Air Intelligence Center (NAIC) at Wright-Patterson AFB, OH.

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Vita

Captain Dwight D. Fullingim was born 19 June 1968 in Beaumont, Texas. He graduated from Silsbee High School in Silsbee, Texas in June 1986. One month later he entered the United States Air Force Academy Class of 1990. He graduated 30 May 1990 with a Bachelor of Science degree in Astronautical Engineering and was commissioned as a Second Lieutenant in the United States Air Force.

Captain Fullingim's first assignment was to the 86th and 87th Flying Training Squadrons at Laughlin Air Force Base, Del Rio, Texas. He has had two other assignments, the first to the Ballistic Missile Organization at Norton Air Force Base, San Bernardino, California and the second to the National Air Intelligence Center, Wright-Patterson Air Force Base, Dayton, Ohio. He entered the Graduate Systems Engineering program at the Air Force Institute of Technology in May, 1996.

His next assignment is to the Air Force Operational Test and Evaluation Center at Kirtland Air Force Base in Albuquerque, New Mexico.

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Vita

Captain Brian L. James was born on 16 March 1967 in Hunstville, Alabama. He attended Clear Lake High School in Houston, Texas. He graduated from the United States Air Force Academy in 1990, with a Bachelor of Science in Engineering Sciences. The degree specialized in the field of Astrodynamics. Upon graduation from the Academy, he received a full commissioning as a Second Lieutenant in the United States Air Force. In March of 1992, Captain James completed his first assignment at Williams AFB in Phoenix, Arizona. He graduated top in his class from Undergraduate Pilot Training (UPT) and was assigned to fly the F-15 Eagle as his follow-on assignment. After completing the Fighter Lead-In program at Holloman AFB in New Mexico, Captain James was medically disqualified while performing his F-15 training at Tyndall AFB. From 1992 through 1996, he acted as the Intelligence Special Collections Manager for the National Air Intelligence Center (NAIC) at Wright-Patterson AFB in Dayton, Ohio. In May of 1996, he entered the School of Engineering, Air Force Institute of Technology. Captain James' follow-on assignment is to the Air Force Operational Test and Evaluation Center, Kirtland AFB, New Mexico.

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Vita

Captain James Valenti was born on 14 February 1968 in Pittsburgh, PA. He graduated from Penn Hills High School in Penn Hills, PA June 1986. In the Fall of 1986, he entered Penn State University, and graduated in May 1990 with a Bachelor of Science degree in Aerospace Engineering. Upon graduation, he was commissioned as a Second Lieutenant and was an AFROTC Distinguished Graduate.

After completing Undergraduate Space Training at Lowry AFB, CO, his first assignment was to Offutt AFB, NE. There, he worked as a Satellite Crew Commander for the 6th Space Operations Squadron (6 SOPS). He was responsible for the command and control of Defense Meteorological Satellite Program (DMSP) satellites. He also served as a Satellite Flight Commander and Satellite Engineer with the 6 SOPS. His second assignment was also at Offutt AFB, NE, where he served as the DMSP System Program Office's Customer Liason Officer. In May 1996, he entered the Graduate Space Operations program in the Air Force Institute of Technology's School of Engineering.

His next assignment is to the Air Force Technical Applications Center (AFTAC) at Patrick AFB, FL.

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Vita

Captain Jörg D. Walter was born on 24 July 1969 in Seattle, Washington. He graduated from Pioneer High School in Ann Arbor, Michigan and entered undergraduate studies at Michigan State University in East Lansing, Michigan. He graduated with a Bachelor of Science degree in Mechanical Engineering in March 1992. While at Michigan State he enrolled in Air Force ROTC and received his commission on 21 March 1992.

His first assignment was at Wright-Patterson AFB as a software program manager for the Advanced Tactical Airborne Reconnaissance System (ATARS). He next served as a project officer for the Commercial Aircraft Program Office and worked on the C-20H, C-32, C-37 and YAL-1A programs. In May 1996, he entered the School of Engineering, Air Force Institute of Technology.

Captain Walter's follow-on assignment is to the Air Force Operation Test & Evaluation Center, Kirtland AFB, New Mexico.

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Vita

First Lieutenant Paul J. Cotter, Jr. was born on 30 July 1972 in Altoona, PA. He graduated from Altoona High School in Altoona, PA and entered undergraduate studies at The Pennsylvania State University in University Park, PA. He graduated with a Bachelor of Science degree in Aerospace Engineering in May 1992. He enrolled in Air Force Officer Training School in September 1993 and received his commission 15 March 1994.

His first assignment was at Whiteman AFB as a squadron adjutant for the 509th Operations Support Squadron. In May 1996, he entered the School of Engineering, Air Force Institute of Technology.

Lieutenant Cotter's follow-on assignment is to the 45th Range Squadron at Cape Canaveral, Florida.

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Vita

First Lieutenant William A. Seeliger was born on 20 Aug 1969 in Houston, Texas. He graduated from Nimitz High School in 1987 and entered undergraduate studies at North Harris College in Houston, Texas in the same year. He received an Associate of Arts degree in 1990. He continued his undergraduate education at Texas A&M University in College Station, Texas and graduated with a Bachelor of Science degree in Physics in 1993. He entered the U.S. Air Force Officer Training School on 25 Jan 1994 and received his commission on 03 May of the same year.

His first assignment was at Kirtland AFB in Albuquerque, New Mexico for scientific and engineering research at the U.S. Air Force Phillips Laboratory. In May 1996 he began graduate studies at the Air Force Institute of Technology's Graduate School of Engineering at Wright-Patterson AFB in Dayton, Ohio.

Lieutenant Seeliger's follow-on assignment is to the 55th Space Weather Squadron at Falcon AFB in Colorado Springs, Colorado.

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